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PROGRAM APOLLO FLIGHT MISSION DIRECTIVE
APOLLO MISSION A-003
(BP-22) (U)

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
Houston, Texas

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NASA Program Apollo Working Paper No. 1145A
PROGRAM APOLLO FLIGHT MISSION DIRECTIVE
APOLLO MISSION A-003
(BP-22)

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Houston, Texas
April 16, 1965

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1.0 INTRODUCTION

1.1 General Mission Objectives

Mission A-003 utilizes Boilerplate 22 (BP-22) spacecraft and Little Joe II launch vehicle 12-51-2. There are two major objectives of Apollo Mission A-003:

(a) To demonstrate launch escape vehicle performance at an altitude approximating the upper limit for the canard subsystem.

(b) To demonstrate orientation of the launch escape vehicle to a "main heat shield forward" attitude after abort.

1.2 Mission Directive

This Mission Directive supplements the Apollo Program General Test Plan (reference 1) and the Apollo Test Requirements (reference 2), and takes precedence over all other documents to control the test and to provide the necessary information for all phases of test planning. The White Sands Missile Range (WSMR) facilities, logistics support, and range contractor requirements are defined in the Operations Requirements Document (reference 3).

1.3 Revisions

This Mission Directive will be revised to reflect major changes in test plans or philosophy.

2.0 TEST OBJECTIVES AND SUBSYSTEM PRIORITIES

2.1 Mission A-003 Test Objectives

2.1.1 First-order test objectives.- The first-order test objectives are:

(a) Demonstrate satisfactory launch escape vehicle (LEV) performance at an altitude approximating the upper limit for the canard subsystem.

(b) Demonstrate orientation of the LEV to a main heat shield forward attitude after abort.

2.1.2 Second-order test objectives.- The second-order test objectives are:

(a) Determine the damping of LEV oscillations with the canard subsystem deployed.

(b) Demonstrate the separation of the Launch Escape Subsystem (LES) plus Boost Protective Cover (BPC) by the tower jettison motor, and jettison of the forward heat shield by the thrusters.

(c) Determine degradation in window visibility due to rocket motor exhaust products for an abort in the region of abort mode transition altitude.

2.1.3 Third-order test objectives.- The third-order test objectives are:

(a) Determine the physical behavior of the boost protective cover during launch and entry from high altitude.

(b) Obtain data on thermal effects during boost and during impingement of the launch escape motor plumes on the command module and the launch escape tower.

(c) Determine pressures on the command module boost protective cover during launch and high altitude abort.

(d) Demonstrate performance of the earth landing subsystem using the two-point harness attachment for the main parachutes.

(e) Determine vibration and acoustic environment and response of the service module with simulated reaction control subsystem motor quadrants.

2.2 Subsystem Priorities

2.2.1 Launch vehicle.- The priorities of the launch vehicle subsystems are:

(a)	Algol motors	Primary
(b)	Airframe	Primary
(c)	Launcher	Primary
(d)	Electrical power subsystem	Primary
(e)	Attitude control subsystem	Primary
(f)	Range safety subsystem	Primary
(g)	RF command subsystem	Primary

2.2.2 Spacecraft.-

2.2.2.1 Launch escape subsystem: The priorities of the launch escape subsystem components are:

(a)	Launch escape motor	Primary
(b)	Tower jettison motor	Primary
(c)	Pitch control motor	Primary
(d)	Tower release mechanism	Primary
(e)	Tower structure	Primary
(f)	Canard subsystem	Primary

2.2.2.2 Earth landing subsystem: The priority of the earth landing subsystem is:

(a)	Parachutes and deployment equipment	Primary

2.2.2.3 Electrical power subsystem: The priorities of the electrical power subsystem components are:

(a)	Batteries and distribution subsystem	Primary
(b)	Earth landing subsystem sequencer	Primary
(c)	Mission sequencer	Primary

2.2.2.4 Communication and instrumentation subsystem: The priorities for the communication and instrumentation subsystem components are:

(a)	Telemetry subsystem	Primary
(b)	Onboard recorders (2)	Primary
(c)	End instruments and signal conditioners	See Note 1
(d)	C-Band radar beacons	See Note 2
(e)	Onboard cameras	See Note 3

2.2.2.5 Structure subsystem: The priorities of the structure subsystem are:

(a)	Command module structure	Primary
(b)	Service module structure	Primary
(c)	Adapter structure	Primary
(d)	Command module-to-service module release mechanism	Primary

Note 1 - Priority assignments, made for planning purposes, are shown in reference 4. The final assignment of priorities will be made in the Mission Rules, reference 12.

Note 2 - One of the two C-Band radar beacons is primary.

Note 3 - Of the three cameras, only the launch escape tower camera is primary.

(e)	Forward heat shield separation thrusters	Primary
(f)	Boost protective cover	Primary

2.2.3 Ground-based support systems.- The priorities of the ground-based support systems are:

(a)	Radar tracking (WSMR range safety)	Primary
(b)	Optical tracking	Primary
(c)	Telemetry stations	Primary
(d)	Range meteorological network	Primary
(e)	Range timing network	Primary
(f)	Range communication network	Primary
(g)	Real-time data system	Secondary
(h)	Command destruct signal generator	Primary

3.0 MISSION A-003 DESCRIPTION

3.1 General Flight Test Plan

The test vehicle will be launched from the White Sands Missile Range (WSMR), the altitude of which is approximately 4,000 feet above mean sea level (msl). The flight path will be approximately 5 degrees West of true North. Figure 3-1 illustrates the launch area, flight path, and landing areas at WSMR.

All phases of Mission A-003 are illustrated in figure 3-2. These phases are:

(a) Acceleration of the test vehicle to the test point by the Little Joe II launch vehicle.

(b) Abort of the launch escape vehicle including: launch escape motor burning, pitch control motor burning, and deployment of the canard surfaces.

(c) Coast of the launch escape vehicle to trajectory apogee, and return to an altitude where atmospheric density is sufficient for the canard surfaces to properly orient the launch escape vehicle.

(d) Flight of the launch escape vehicle stabilized by the canards, launch escape subsystem (LES) jettison, and flight of command module stabilized by the dual drogue parachutes.

(e) Landing of the command module by parachute.

Ballistic flight of the Little Joe II and service module after separation from the launch escape vehicle is also shown in figure 3-2.

Six Algol rocket motors power the Little Joe II launch vehicle -- no Recruit motors are used for this mission. At launch, a signal transmitted from the blockhouse to the launch vehicle via landline ignites three motors. After approximately 40 seconds, when the first three motors burn out, the remaining three motors ignite; after burnout of the last three motors, a radio command signal from the ground initiates the abort sequence.

The abort sequence is initiated within the Mach number, dynamic pressure, and altitude parameters defined by the test point envelope in figure 3-3.

If the radio command fails to initiate the abort sequence, a backup timer in the CM is programed to initiate the abort sequence. A range safety subsystem, controlled by the WSMR range safety officer, can also be used to terminate launch vehicle thrust and initiate the abort sequence if necessary.

The abort signal initiates separation of the command module from the service module, ignition of the pitch control and launch escape motors, and activation of the mission sequencer 11-second time delay. Eleven seconds after abort initiation, the canards are deployed and locked in the open position. During the tumbling coast phase of the trajectory, the launch escape vehicle reaches an apogee of approximately 175,000 feet msl.

As the launch escape vehicle reenters the atmosphere, the effectiveness of the canards increases until tumbling stops and the launch escape vehicle stabilizes with the aft heat shield forward. At approximately 25,000 feet, the launch escape subsystem and boost protective cover are jettisoned as a unit by the tower jettison motor and, 0.4 seconds later, the forward heat shield is jettisoned by thrusters. This sequence is initiated by a baroswitch.

The dual drogue parachutes are deployed in a reefed condition two seconds after the launch escape subsystem is jettisoned and are dereefed eight seconds after deployment. At approximately 12,000 feet, a second baroswitch initiates separation of the dual drogue parachutes from the command module and ignition of the three pilot parachute mortars. The pilot parachutes extract the main parachutes, which remain in a reefed condition. Eight seconds later, the main parachutes are dereefed and become fully inflated. The rate of descent is reduced to approximately 26 feet per second by the main parachutes and, approximately 9 minutes after lift-off, the command module lands.

3.2 Detailed Test Sequence

The mission trajectory parameters are listed in table 3-I and the sequence of events is illustrated in figure 3-4.

3.3 Trajectories

Predicted trajectories for Mission A-003 are shown in figures 3-1, 3-3, 3-5, and 3-6.

TABLE 3-I.- MISSION A-003 NOMINAL TRAJECTORY PARAMETERS

Event	Time, seconds	Mach number (Note 2)	Dynamic pressure, psf (Note 2)	Altitude (msl), feet	Range, feet (Note 1)	Flight path angle, deg
Lift-off	T + 0	0	0	3,989	0	84.0
Abort	T + 89.0	3.708	154.0	111,176	96,484 N. 8,728 W.	36.92
Canard deployment	T + 100.0	3.628	60.3	133,006	135,050 N. 12,223 W.	26.2
Peak altitude	T + 151.36	2.922	7.8	175,337	301,667 N. 27,295 W.	0.0
Tower jettison	T + 294.6	0.542	157.4	26,638	565,349 N. 49,237 W.	-83.8
Drogue parachute deployment	T + 296.6	0.482	130.0	25,593	565,425 N. 49,249 W.	-84.56
Main parachute deployment	T + 351.4	0.192	36	11,591	565,364 N. 49,259 W.	-89.95
CM touchdown	T + 536.7	0.024	0.7	5,000	565,186 N. 49,232 W.	-90.0

Note 1: N. = North, W. = West

Note 2: The maximum dynamic pressure of 892 psf occurs at T + 55 seconds and the maximum Mach number of 4.17 occurs at T + 91.5 seconds.

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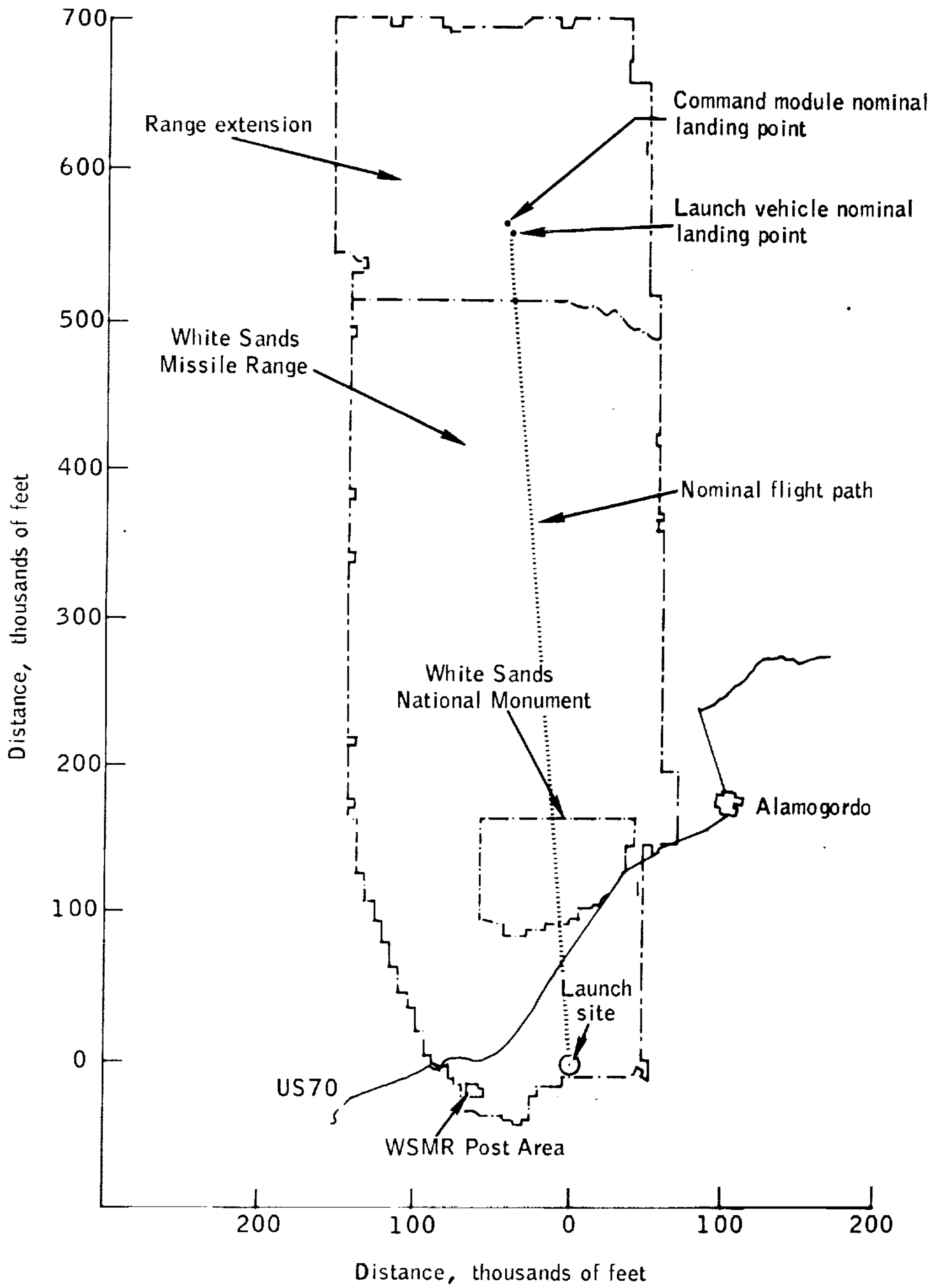

Figure 3-1. - Mission A-003 launch and landing areas at WSMR

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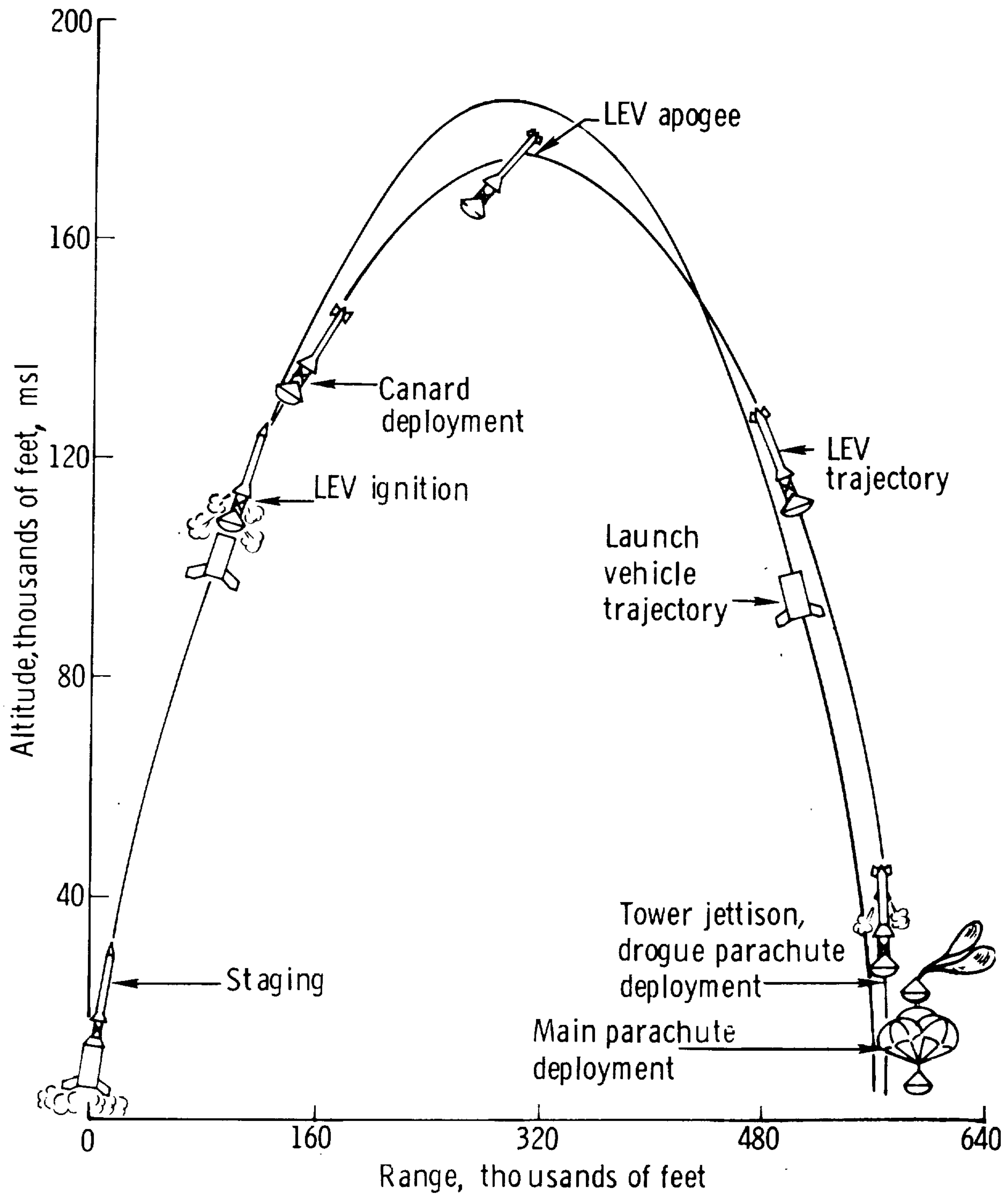

Figure 3-2. - Mission A-003 nominal mission profile

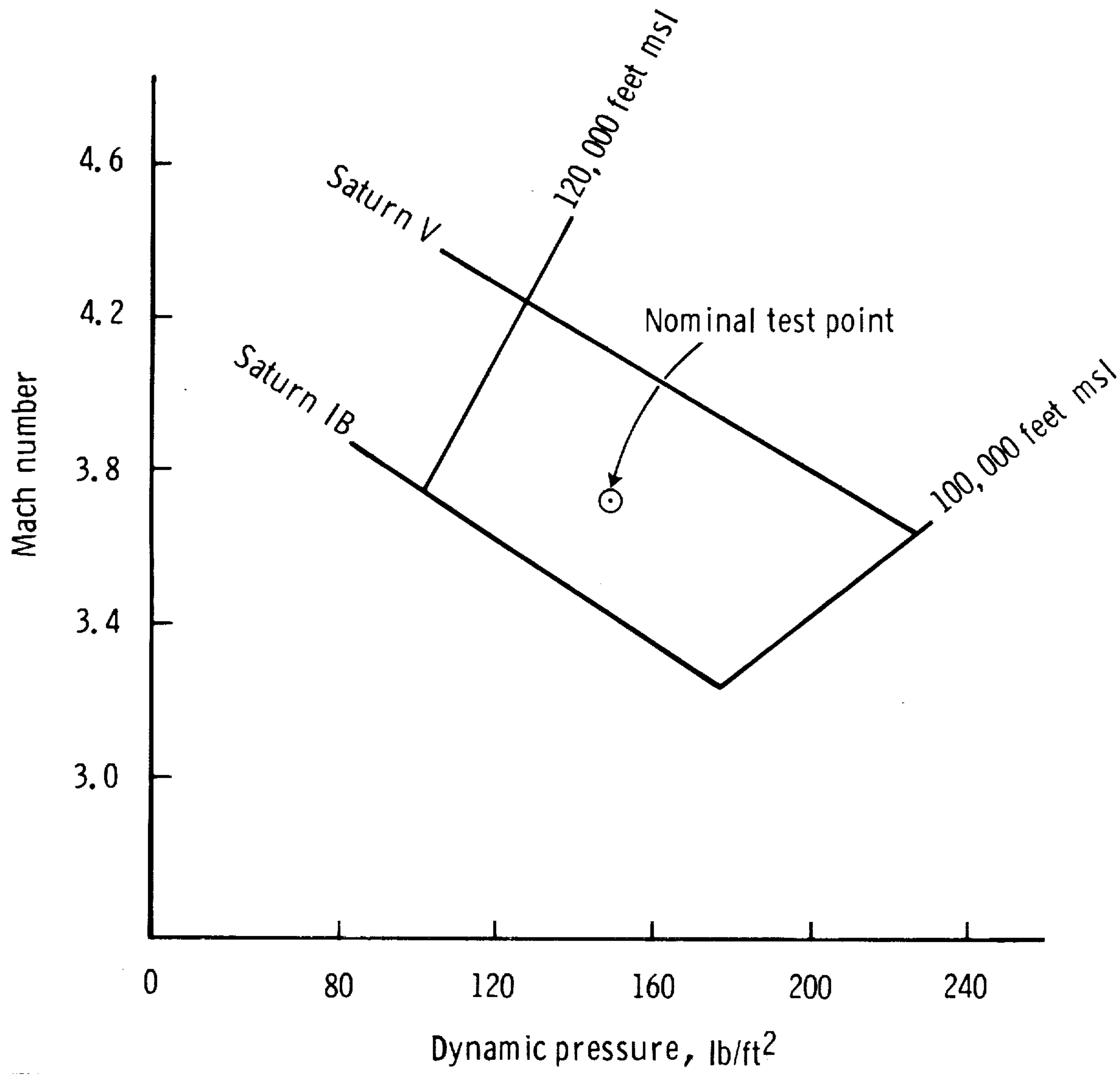

Figure 3-3. - Mission A-003 test point envelope (Mach number as a function of dynamic pressure)

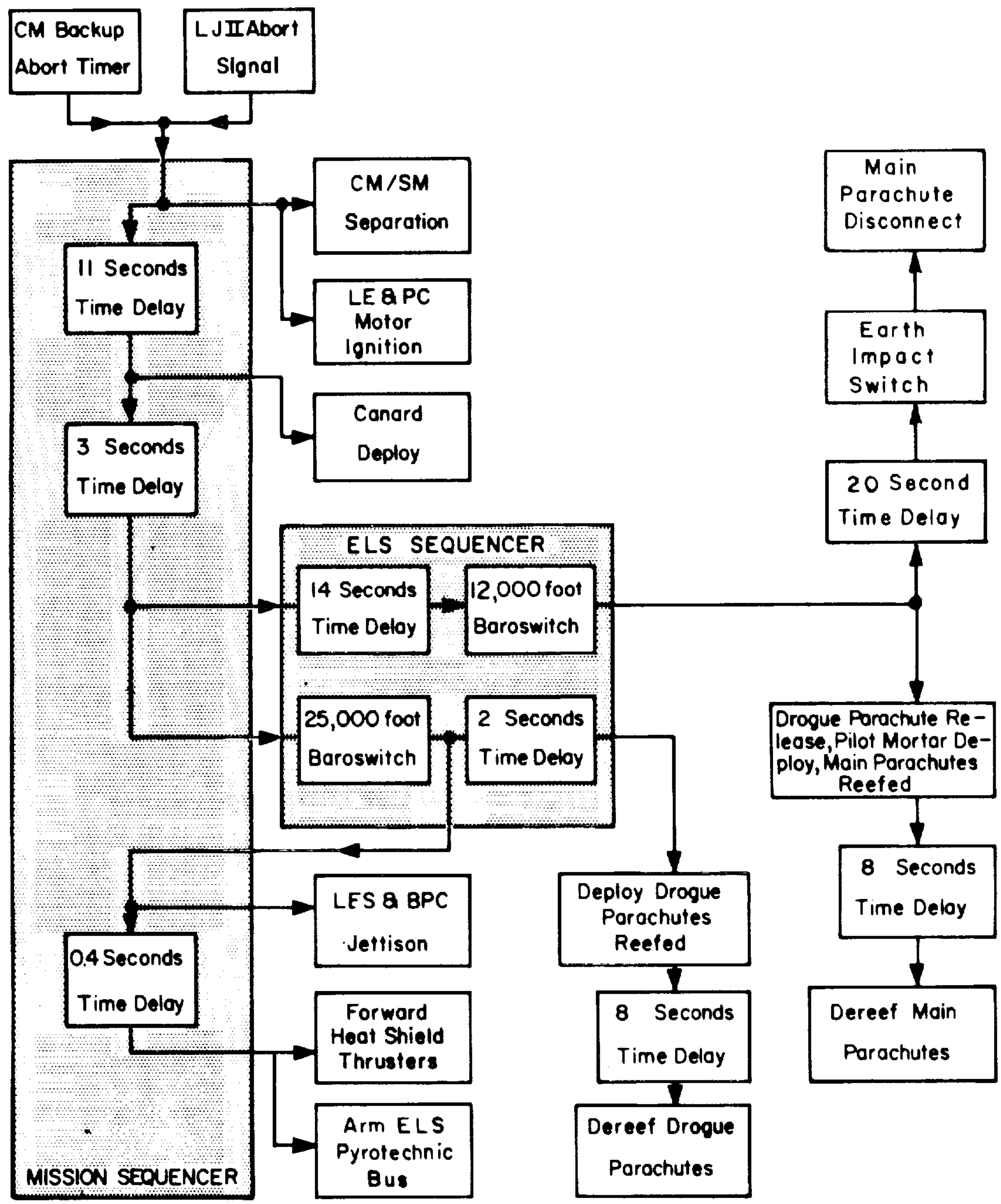

Figure 3-4. - Mission A-003 sequence of events block diagram

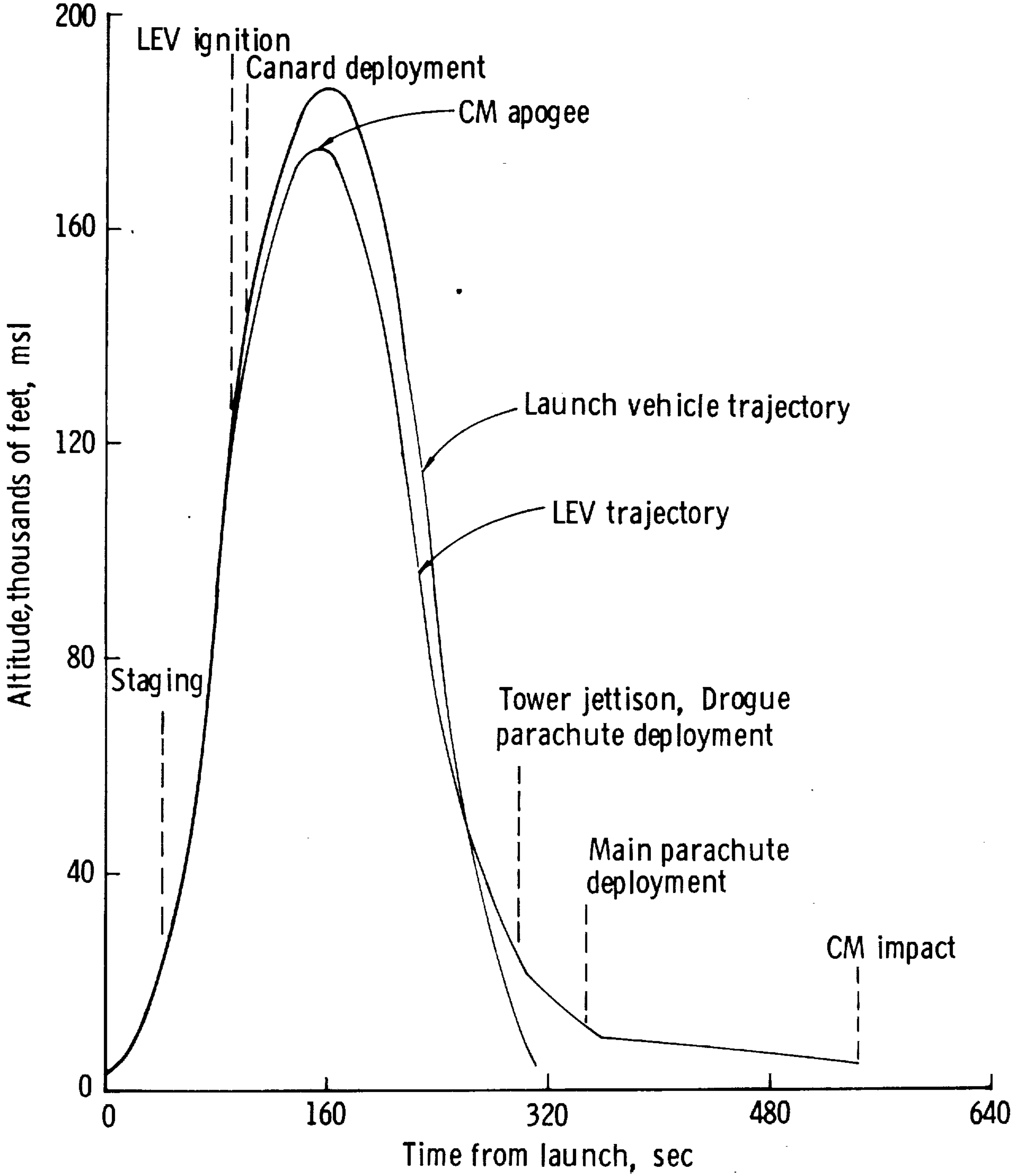

Figure 3-5. - Mission A-003 predicted nominal trajectories (altitude as a function of time)

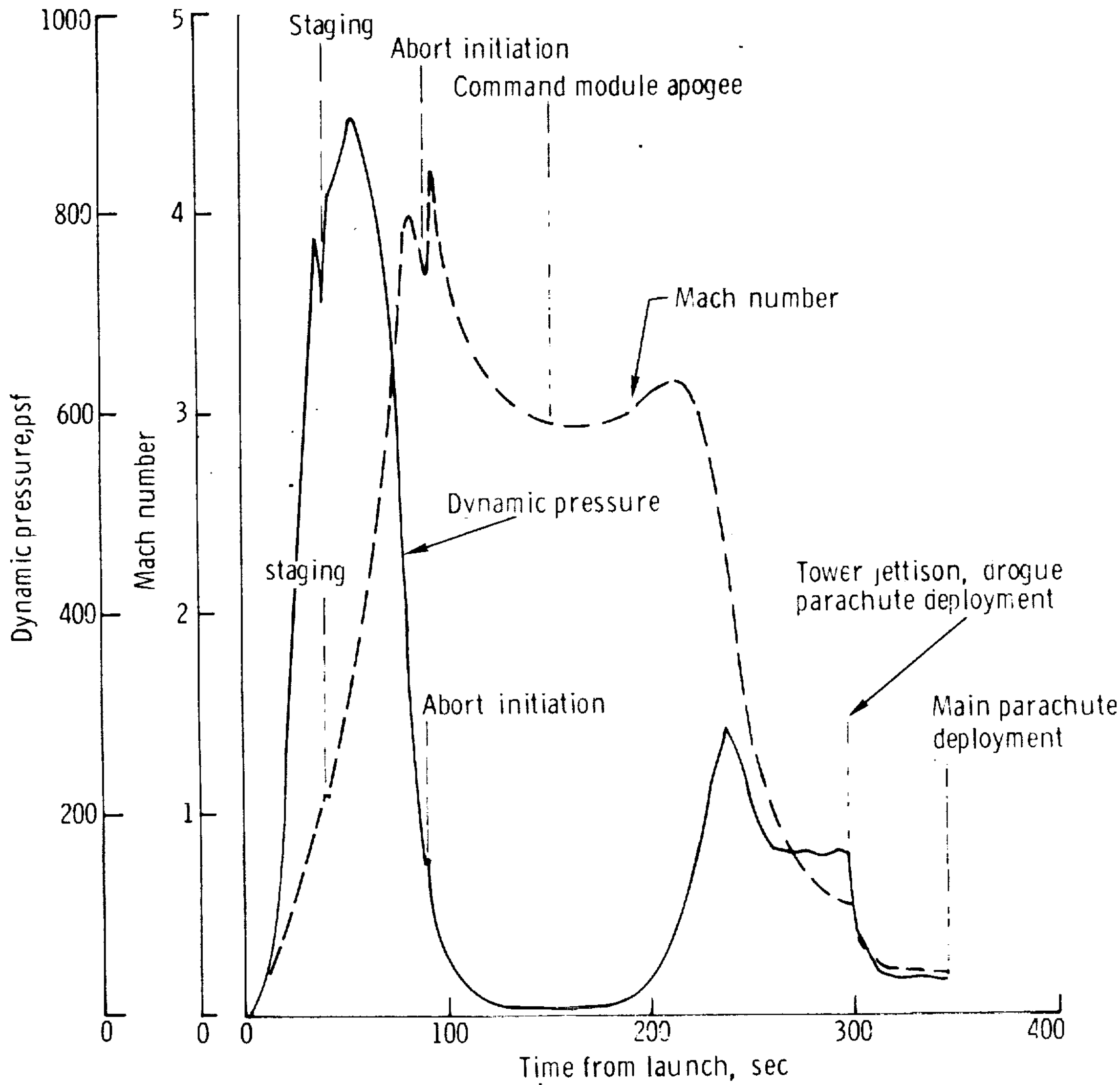

Figure 3-6. - Mission A-003 predicted nominal trajectories (dynamic pressure and Mach number as a function of time)

4.0 TEST VEHICLE DESCRIPTION

4.1 General

The test vehicle for Mission A-003 includes the Boilplate 22 (BP-22) spacecraft and the Little Joe II launch vehicle (figures 4-1, 4-2, and 4-3). The BP-22 spacecraft is composed of boilerplate command and service modules, and a launch escape subsystem. Together, the launch escape subsystem and the command module are known as the launch escape vehicle (figure 4-4).

The test vehicle for Mission A-003 is basically the same as the vehicle used for Mission A-002 (BP-23, reference 5); however, there are significant differences. The major differences are indicated in table 4-I.

4.2 Test Vehicle Weight and Balance Summary

Calculated weights, centers of gravity, and moments of inertia for the Mission A-003 test vehicle are available in reference 6.

4.3 Structural Design Criteria

4.3.1 Spacecraft.- The primary structure of the BP-22 spacecraft is designed to withstand the expected environmental conditions. Details are available in reference 7.

4.3.2 Launch vehicle.- The structure of the Little Joe II launch vehicle is designed to carry payload weights up to 80,000 pounds and is capable of withstanding an angle of attack-dynamic pressure value of 10,000 deg-lb/ft^2. The ratio of design limit load to design ultimate load is 1.5. Additional launch vehicle design information is available in reference 8.

4.4 Command Module Description

The boilerplate command module (figure 4-4) is a conical structure approximately 135 inches high and 154 inches in diameter at the base. Command module components are described in the following paragraphs.

4.4.1 Forward bulkhead and egress tunnel.- The egress tunnel is a welded-aluminum tube that is welded to the forward bulkhead. A cover plate, which serves as a camera mount, is bolted to the top of the tunnel. The compartment formed by the upper deck of the forward bulkhead and the egress tunnel is divided into four sections by aluminum stiffeners riveted to the structure. Components of the earth landing subsystem (mortars, parachutes, etc.) are housed in this compartment.

4.4.2 Crew compartment.- The command module shell covering the crew compartment consists of two subassemblies: (1) the forward section and (2) the aft section. The forward section attaches to the forward bulkhead while the aft section attaches to the aft heat shield structure. Equipment bays, brackets, and other secondary structures used to mount the functional subsystem components are included in the crew compartment.

4.4.3 Aft heat shield.- For this flight, the aft heat shield protects the command module from earth landing damage. This semi-spherical structure consists of an aluminum honeycomb core with inner and outer skins of phenolic-impregnated, laminated glass cloth. Aluminum inserts in the shield are used as attach fittings and as compression pads to mate the command module to the service module.

4.4.4 Forward heat shield.- The forward heat shield is a truncated cone that covers the upper third of the command module thus enclosing the earth landing subsystem (ELS), egress hatch, and other equipment. Shortly after the LES and boost protective cover are jettisoned, the forward heat shield is jettisoned by four gas generator thrusters.

4.4.5 Hatches and antennas.- Access to the command module interior is provided by the main hatch, which is located in the sidewall over the head of the center couch position (minus Z axis). Eight openings in the shell of the forward crew compartment are provided for antennas.

4.4.6 Main parachute bridle attach points.- Two fittings, located approximately 180 degrees apart on the top, outside surface of the egress tunnel, are provided for attachment of the main parachute bridle.

4.4.7 Simulated CM-SM umbilical and scimitar antennas.- A simulated CM-SM umbilical and two simulated scimitar antennas are attached to the outer surface of the CM shell.

4.4.8 Boost protective cover.- The boost protective cover (figure 4-5) extends over the entire conical surface of the command module and CM-SM interface. For Saturn-boosted flights, a boost protective cover will provide protection from aerodynamic heating during exit flight. The upper third and the lower CM-SM interface portions of the cover, which consist of ablative cork insulation supported by a glass-honeycomb matrix over a teflon-impregnated glass cloth, are called hard covers. The lower two-thirds of the cover, consisting of ablative cork insulation over a teflon-impregnated glass cloth, is called a soft cover. The hard frame encloses a simulated window located over the main hatch. Other hard parts, shaped to enclose the protuberances formed by the simulated CM-SM umbilical and scimitar antennas, are also incorporated in the boost protective cover.

4.4.9 Forward heat shield thruster subsystem.- This subsystem (figure 4-6) consists of two identical subsystems, each composed of a gas generator attached to two thruster units separated by 180°; thus, there are four thruster units spaced 90° apart around the ELS compartment. Each subsystem produces sufficient force to break the tension ties that attach the forward heat shield to the command module.

4.4.10 Earth landing subsystem.- The ELS consists of pyrotechnic and pyrotechnic-actuated devices, two conical-ribbon drogue parachutes with reefing, three pilot parachutes, three open-ring, ring-sail main parachutes, deployment bags, bridles, risers, and an ELS sequencer. Deployed in the reefed condition by mortars, the dual drogue parachutes remain reefed for eight seconds then fully inflate. The dual drogue parachutes have a nominal diameter of 13.7 feet. A drogue disconnect subsystem (figure 4-7) then separates the drogue parachutes from the command module and the mortars that deploy three parachutes are fired. The pilot parachutes, which have a nominal diameter of 7.2 feet, pull the main parachutes from deployment bags. Deployed 11 percent reefed, the main parachutes remain reefed for 8 seconds then fully inflate to approximately 70 percent of their nominal diameter of 83.4 feet.

4.4.11 Mission sequencer.- Activated by the abort initiation signal, the mission sequencer generates signals that initiate three distinct sequences of events:

(a) Separation of the command module from the service module, ignition of the launch escape and pitch control motors, and deployment of the canards.

(b) Activation of the LES jettison motor circuits, activation of the LES-CM separation circuits, and firing of the forward heat shield thrusters (0.4 seconds after LES jettison). (These events occur only after activation of the ELS 25,000 foot baroswitches).

(c) Activation of the ELS sequencer

If the primary abort initiation method (the abort signal from the Little Joe II) fails, a backup timer in the command module is programed to initiate the abort sequence at T plus 91 seconds.

4.4.12 ELS sequencer.- This sequencer, which consists of relays, baroswitches, and timing devices, provides the impulse which initiates the following events:

(a) Ignition of the LES jettison motor and LES-CM separation

(b) Firing of the drogue parachute mortars

(c) Disconnection of the drogue parachutes and firing of the pilot parachute mortars

4.4.13 Command module data acquisition subsystem.- The airborne instrumentation consists of telemetry equipment, cameras, and tape recorders used to acquire inflight data. A description of this equipment is included in Section 6.0.

4.4.14 Electrical power subsystem.- The electrical power subsystem (figure 4-8) consists of six batteries connected to power distribution busses, wiring, and switches to provide electrical power to the spacecraft subsystems. Battery characteristics are listed in table 4-II.

The batteries are connected as follows:

(a) Each of the 2 instrumentation batteries is connected to an independent, non-redundant instrumentation buss.

(b) Each of the 2 logic batteries is connected to an independent, redundant logic buss.

(c) Each of the 2 pyrotechnic batteries is connected to an independent, redundant pyrotechnic buss.

4.5 Launch Escape Subsystem Description

The launch escape subsystem (figures 4-2, 4-4, and 4-9) consists of the following major structures and hardware:

(a) Tower structure

(b) Launch escape motor

(c) Pitch control motor

(d) Tower jettison motor

(e) Explosive bolts (4)

(f) Canard subsystem

(g) Q-ball assembly

(h) Ballast enclosure and ballast

4.5.1 Tower structure.- The launch escape tower, which connects the launch escape motor to the command module, is approximately 120 inches long and 40 by 50 inches wide at the base. The tower legs are attached to the command module by four explosive bolts.

4.5.2 Launch escape motor.- The launch escape motor is a solid propellant motor, 183 inches long and 26 inches in diameter, with four nozzles; each nozzle is canted outward 35° from the longitudinal axis of the motor. A flange is provided at the aft end of the motor allowing the motor to be bolted to the launch escape tower structural skirt.

The thrust vector is offset by approximately 2-3/4 degrees from the motor longitudinal axis by appropriate sizing of the nozzle throat areas. Final alinement of the thrust vector is performed by adjusting the bolts that connect the launch escape tower to the structural skirt.

A pyrogen-type igniter, using redundant hot wire initiators, ignites the motor. Nominal thrust is 155,000 pounds. Theoretical thrust variation with time of the launch escape motor is illustrated in figure 4-10.

4.5.3 Pitch control motor.- The pitch control is a solid propellant motor, 9 inches in diameter and 22 inches in length. The motor housing forms the structure between the Q-ball assembly and the forward end of the jettison motor; the canards are an integral part of the motor housing. To produce a positive pitching movement for lateral displacement of the LEV from the SM-launch vehicle during abort, the total motor impulse of 1,750 lb/sec is directed laterally in the negative Z direction. Theoretical thrust variation with time for the motor is shown in figure 4-11.

4.5.4 Tower jettison motor.- The tower jettison motor is a solid propellant motor, 26 inches in diameter and 47 inches in length. A flange is included at the aft end, allowing the jettison motor to be bolted to the launch escape motor. The jettison motor has two nozzles; each canted 30° from the longitudinal axis of the launch escape motor. The throats of the nozzles have different areas, resulting in the thrust vector offset of about 2-3/4 degrees from the launch escape motor longitudinal axis. Figure 4-12 illustrates the theoretical thrust variation with time.

4.5.5 Explosive bolts.- Four single-mode explosive bolts, one bolt located in each of the tower feet (figure 4-13), release the tower from the CM. Each bolt contains a single explosive charge and incorporates a dual-ignition feature (two hot-wire initiators) to increase reliability. In normal operation both hot-wire initiators fire.

4.5.6 Canard subsystem.- The canard subsystem (figure 4-14) is mounted in the pitch control motor housing. Deployed by two pyrotechnic-actuated, gas generator piston assemblies 11 seconds after the LES jettison motor is ignited, the canards maneuver the CM to the aft heat shield forward position and reduce oscillations during LES descent.

4.5.7 Q-ball assembly.- The Q-ball assembly (figure 4-15) is a cone-shaped unit bolted to the forward end of the pitch control motor housing. Dynamic pressure, angle of attack, and angle of sideslip are obtained by means of a dynamic pressure pickup and four static pressure pickups located 90° apart on the forward end of the Q-ball assembly. This information is used for test vehicle trajectory analysis and evaluation.

4.5.8 Ballast.- The LES ballast compartment is located between the pitch control motor housing and the canard assembly housing. Ballast required to achieve the desired LEV dynamic characteristics during the abort sequence is installed in this compartment.

4.6 Service Module Description

The boilerplate service module is a cylinder 161.75 inches long and 154 inches in diameter. Provisions for bolting the service module to the launch vehicle are incorporated in a 10-inch long adapter placed between the launch vehicle and service module.

A prototype CM-SM separation mechanism is connected to the command module attachment lugs. This subsystem consists of three tension ties located at approximately equal intervals around the periphery of the CM base. Each tie (figure 4-16) includes a flat plate with a shaped charge attached to each side; either charge will sever the plate. Detonators, on separate electrical circuits, are installed at each end of the shaped charges.

The service module does not have a functioning reaction control subsystem (RCS); however, simulated RCS motor quadrants are mounted so that the aerodynamic shape of the BP-22 service module is the same as a Block I service module.

4.7 Little Joe II Launch Vehicle Description

The launch vehicle airframe consists of a cylindrical forebody shell approximately 19 feet long, an afterbody shell approximately 10 feet long, and four fins. Both body sections are 154.0 inches in diameter. The fins are each 50 square feet in area with leading edges swept back 45 degrees relative to the body centerline. Each fin includes a hydraulically-controlled, movable elevon and a reaction control subsystem to control the attitude of the launch vehicle.

A large, built-up bulkhead, called the thrust bulkhead, at the vehicle base is the main structural member of the vehicle. This bulkhead is essentially a pyramid type structure, approximately 21 inches thick at the periphery and 60 inches thick at the center, and consists of rocket motor housing tubes, upper and lower face plates, cylindrical interconnected members, launcher fittings, and fin attach fittings.

4.7.1 Electrical power subsystem.- The electrical power subsystem consists of two primary batteries for vehicle electrical power, and four batteries (two receiver and two pyrotechnic) for the RF command subsystem.

4.7.2 Instrumentation subsystem.- The launch vehicle instrumentation subsystem is described in paragraph 6.2.

4.7.3 Propulsion subsystem.- The propulsion subsystem consists of six Algol ID, Mod I motors bolted to retaining rings in the thrust bulkhead. The bulkhead at vehicle station 34.75 of the forebody provides lateral support for the motors.

4.7.3.1 Algol rocket motors: The Algol ID, Mod I motors are solid propellant motors. The motor nominal thrust at 70°F is shown in figure 4-17.

4.7.3.2 Motor arrangement and firing sequence: Figure 4-3 illustrates the spacing of the Algol motors. Three of the motors are ignited by ground electrical power, and when the motors burnout, the remaining 3 motors are ignited by the launch vehicle electrical power.

4.7.3.3 Propulsion ignition subsystem: The propulsion ignition subsystem distributes launch vehicle electrical power for motor ignition. To launch the Little Joe II vehicle, an electrical pulse from the blockhouse activates the primary ignition timer and a backup timer located in the launch vehicle. These timers distribute launch vehicle power to ignite both first and second stage motors.

4.7.4 Attitude control subsystem.- The Little Joe II launch vehicle attitude control subsystem (figure 4-18) includes two independent reaction control motors and a hydraulically-activated elevon (aerodynamic control surface) incorporated in each of the four fixed fins.

Also included in the attitude control system is an autopilot. By controlling the elevon position and the rate of propellant flow to the reaction control motors, the autopilot determines launch vehicle attitude and pitch rate. Included in the autopilot is a pitch programer, which is set before flight to perform the required time-correlated control functions.

4.7.5 RF command subsystem.- A block diagram of the Little Joe II RF command subsystem is shown in figure 4-19. The RF command subsystems converts coded RF signals, transmitted by FRW-11 ground command transmitters, into the abort initiation signal that activates the command module mission sequencer.

4.7.6 Range safety destruct subsystem.- A destruct subsystem (figure 4-20) has been incorporated in the Little Joe II launch vehicle to provide explosive thrust termination, if necessary. At the initiation of thrust termination by a ground-transmitted signal sent by the WSMR Range Safety Officer, the command destruct subsystem simultaneously activates the command module mission sequencer by generating an abort initiation signal.

4.8 Launcher Description

The launcher is a fabricated steel structure using I-beams for the main supports. Components include: a pivot frame mounted on double-flange, crane-type trucks for rotation to required azimuth; a support platform incorporating pads and pins for vehicle support; screwjacks for tilting the support platform to required elevation angles; and a launcher mast. The mast, attached to the support platform, incorporates two stabilizing support arms for the launch vehicle and a support arm for the spacecraft umbilical harness. Two A-frames attach to the pivot frame and support the platform hinge points.

The launcher is remotely adjustable in both elevation and azimuth. Elevation attitude can be maintained within one-quarter of a degree and azimuth maintained within one-half of a degree. Adjustments can be made for launchings at azimuth angles $\pm 45^\circ$ from some nominal direction and elevations from 75° to 90° vertical. Remote control of the launcher allows adjustments to be made, if necessary, during the countdown with no prolonged holds for manual operations.

TABLE 4-I.- TEST VEHICLE DIFFERENCES

Item	Installed On -		Refer to paragraph-
	Mission A-002 (BP-23 SC)	Mission A-003 (BP-22 SC)	
<u>Spacecraft</u>			
Service module reaction control subsystem (RCS)	None	Non-functional, simulated RCS motor quadrants	4.6
Electrical power subsystem	Seven batteries: 4 pyrotechnic 2 logic 1 instrumentation and ECS	Six batteries: 2 pyrotechnic 2 logic 2 instrumentation	4.4.14
Forward heat shield thruster subsystem	None	Forward heat shield thruster subsystem	4.4.9
Earth landing subsystem	10-foot diameter pilot parachutes, 88.1-foot diameter main parachutes. Main parachute four-strap harness and four-point CM attach Six second reefing time for drogue parachutes and for main parachutes	7.2-foot pilot parachutes, 83.4-foot diameter main parachutes, shorter pilot parachute mortars. Main parachute two-strap harness and two-point CM attach Eight second reefing time for drogue parachutes and for main parachutes	4.4.10

TABLE 4-I.- TEST VEHICLE DIFFERENCES - Concluded

Item	Installed On -		Refer to paragraph-
	Mission A-002 (BP-23 SC)	Mission A-003 (BP-22 SC)	
Command module external configuration	No scimitar antennas or CM-SM external umbilical	Simulated scimitar antennas and simulated CM-SM external umbilical	4.4.7
Boost protective cover	No protuberances	Protuberances to enclose the simulated scimitar antennas and simulated CM-SM external umbilical	4.4.8
Launch vehicle			
Algol rocket motors	2	6	4.7.3
Recruit rocket motors	4	0	
Telemetry subsystem	Yes	No	6.2

TABLE 4-II.- BATTERY CHARACTERISTICS

Quantity	Part number and function	Capacity per battery
2	4095, instrumentation (Note 5)	1500 watt-hours
2	4090, logic (Note 5)	140 watt-hours
2	ME 461-0007-0002, pyrotechnic (Note 6)	(Note 7)

Note 5 NASA, MSC supplied batteries.

Note 6 North American Aviation supplied batteries.

Note 7 This battery is designed to provide a high current surge for a short duration (45 ampere-minutes at 23 volts), therefore the watt-hour capacity is not applicable.

Note: All dimensions in inches

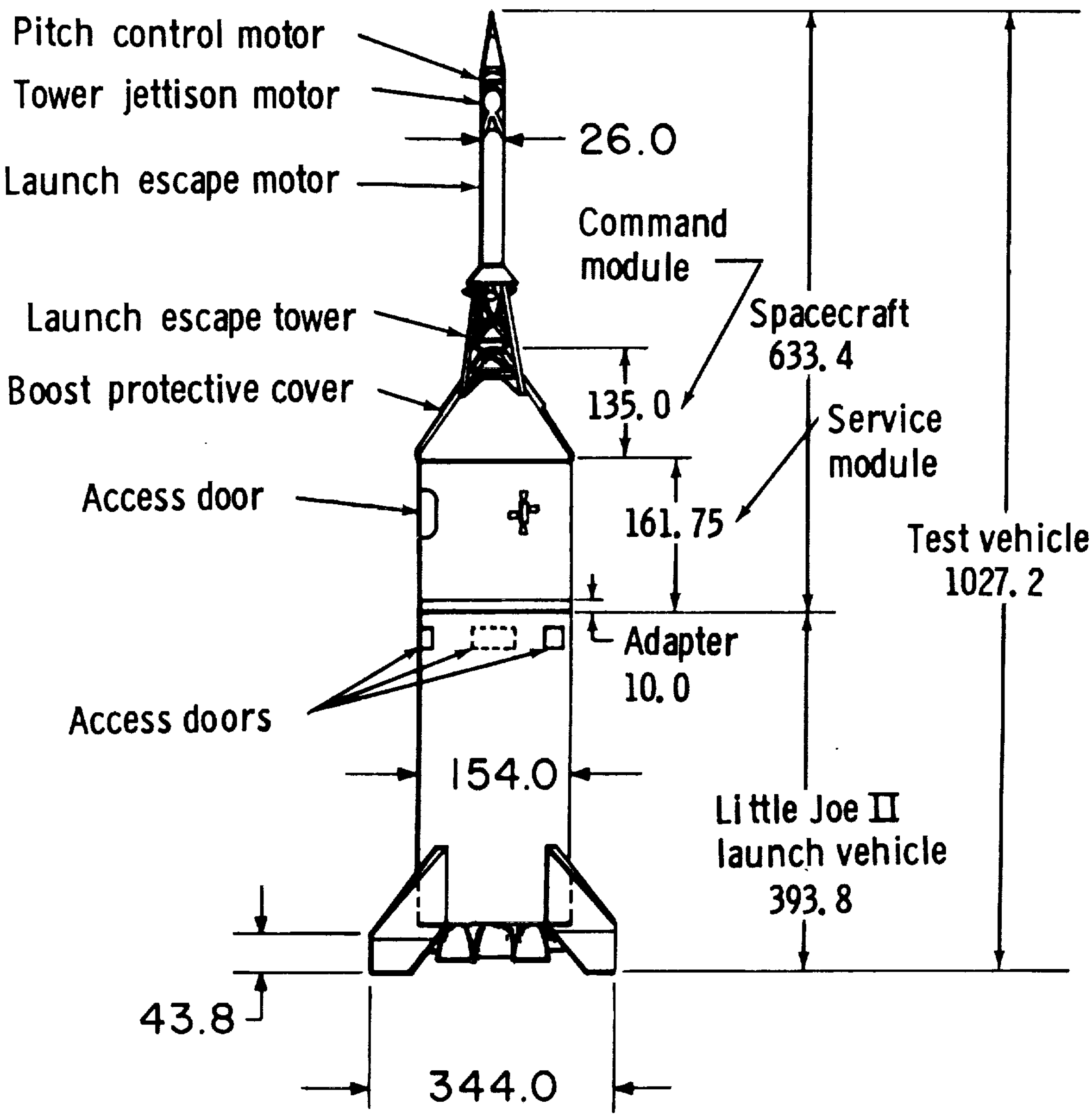

Figure 4-1. - Apollo mission A-003 test vehicle lift-off configuration

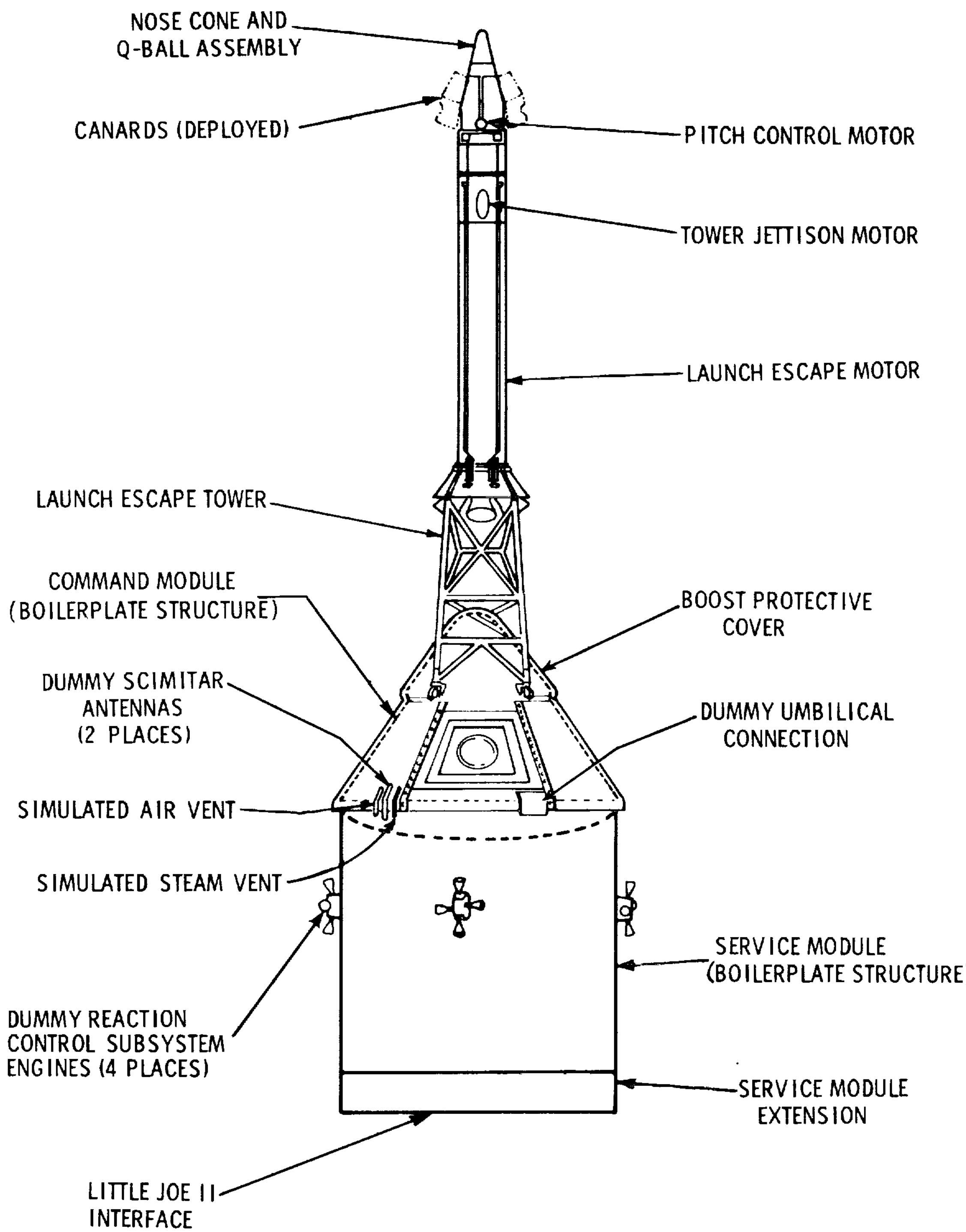

Figure 4-2. - Spacecraft external configuration

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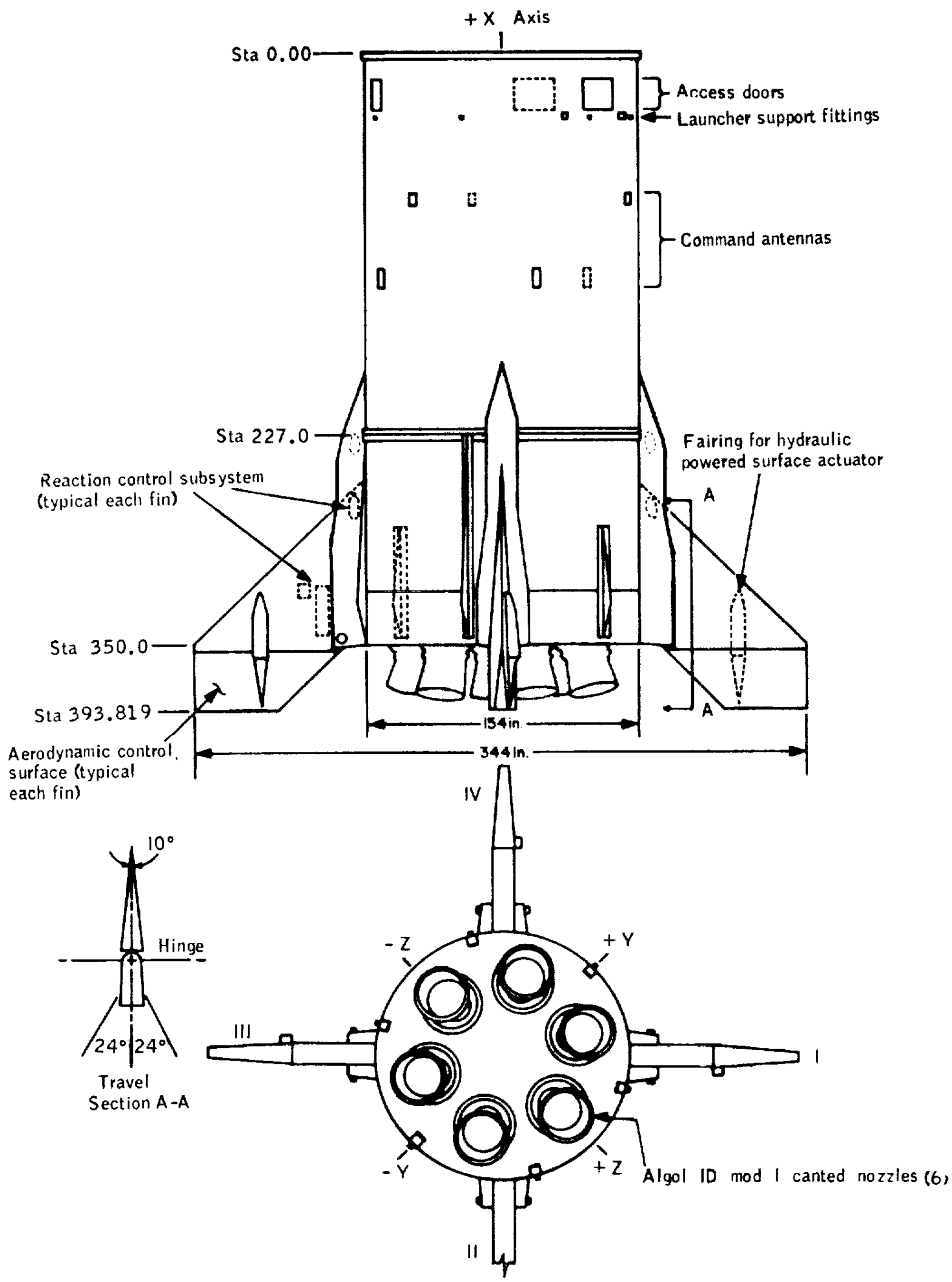

Figure 4-3. - Launch vehicle configuration

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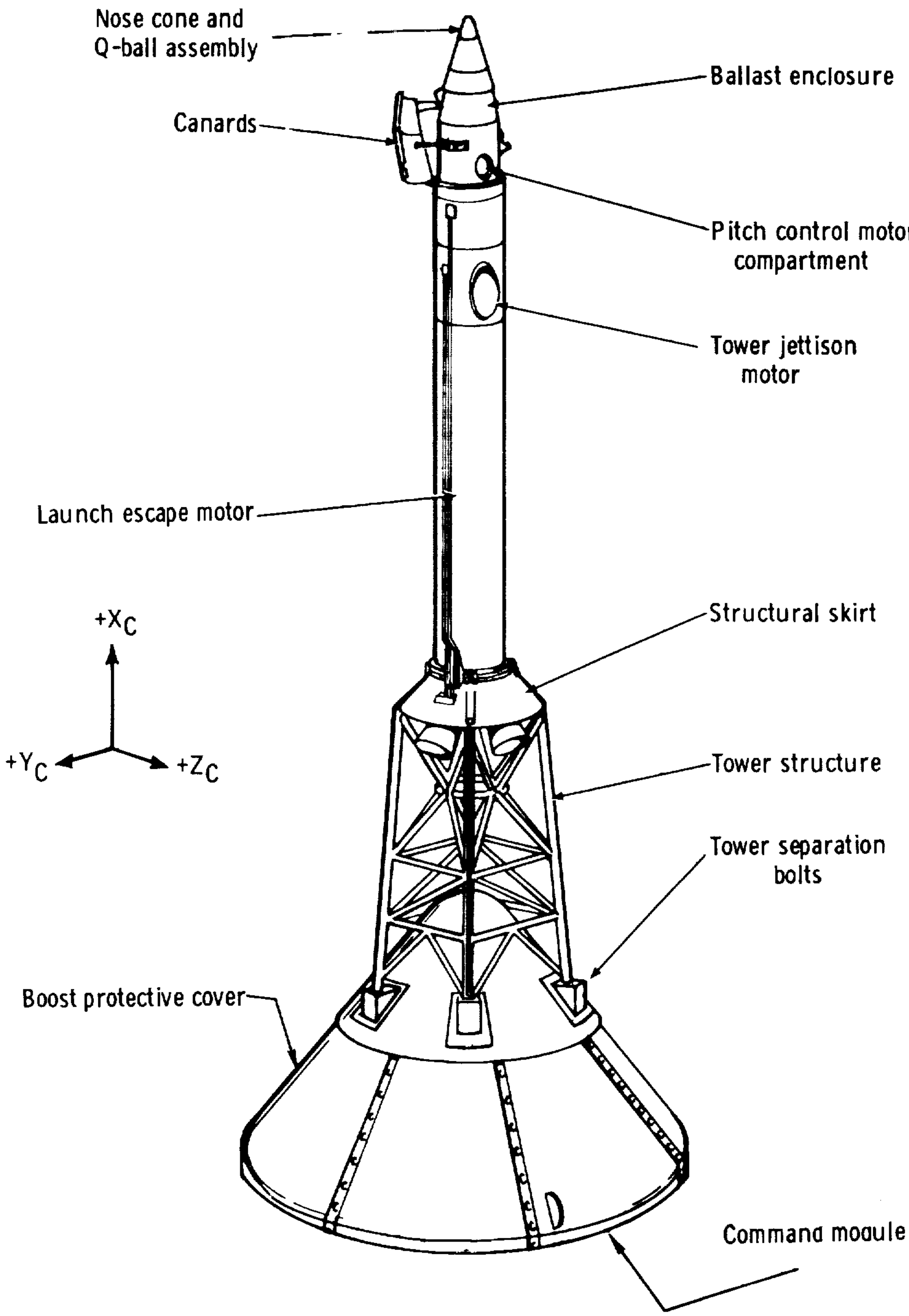

Figure 4-4. - Launch escape vehicle configuration

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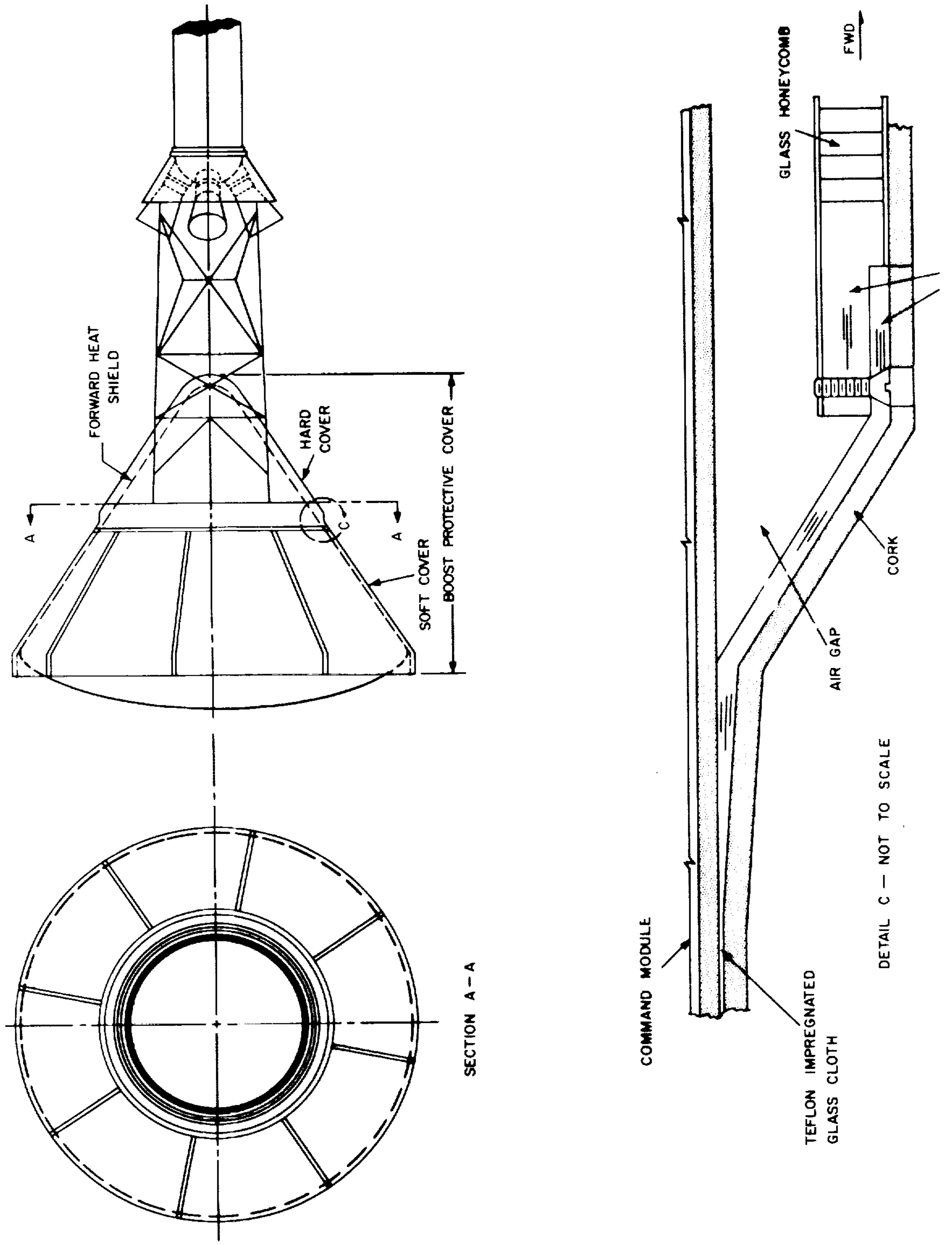

Figure 4-5. - Boost protective cover assembly and details

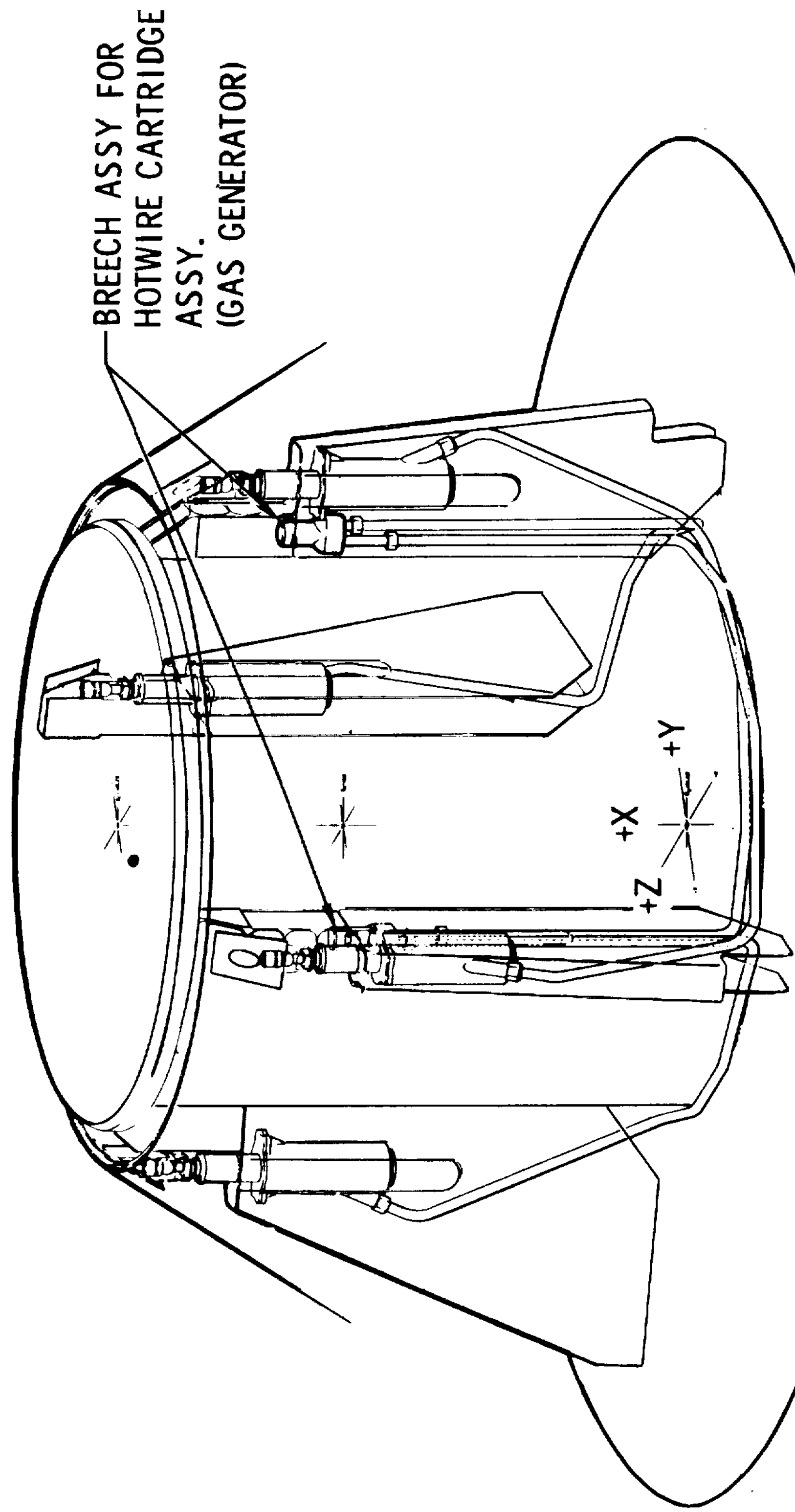

Figure 4-6. - Forward heat shield thruster

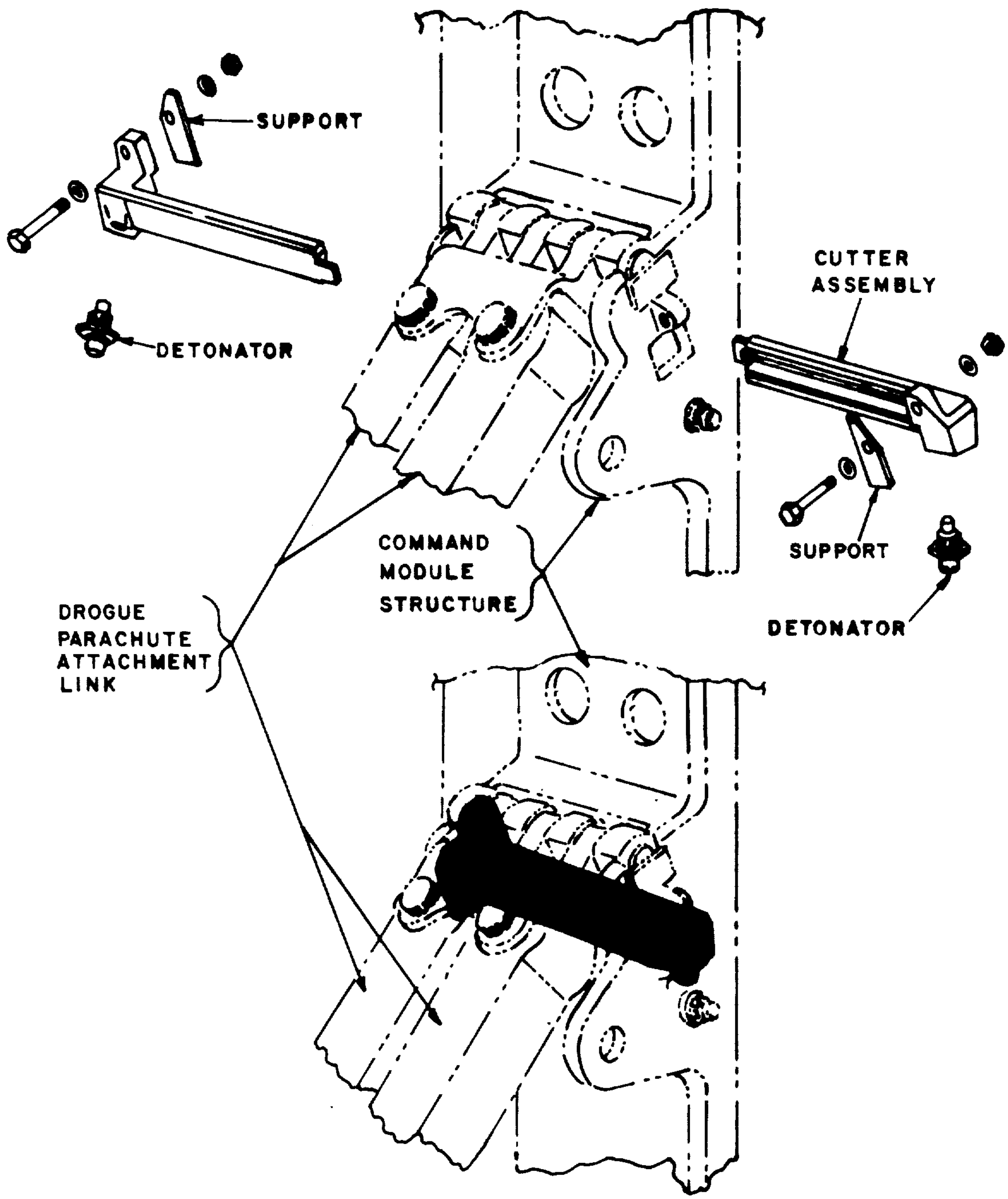

Figure 4-7. - Drogue prarchute disconnect subsystem

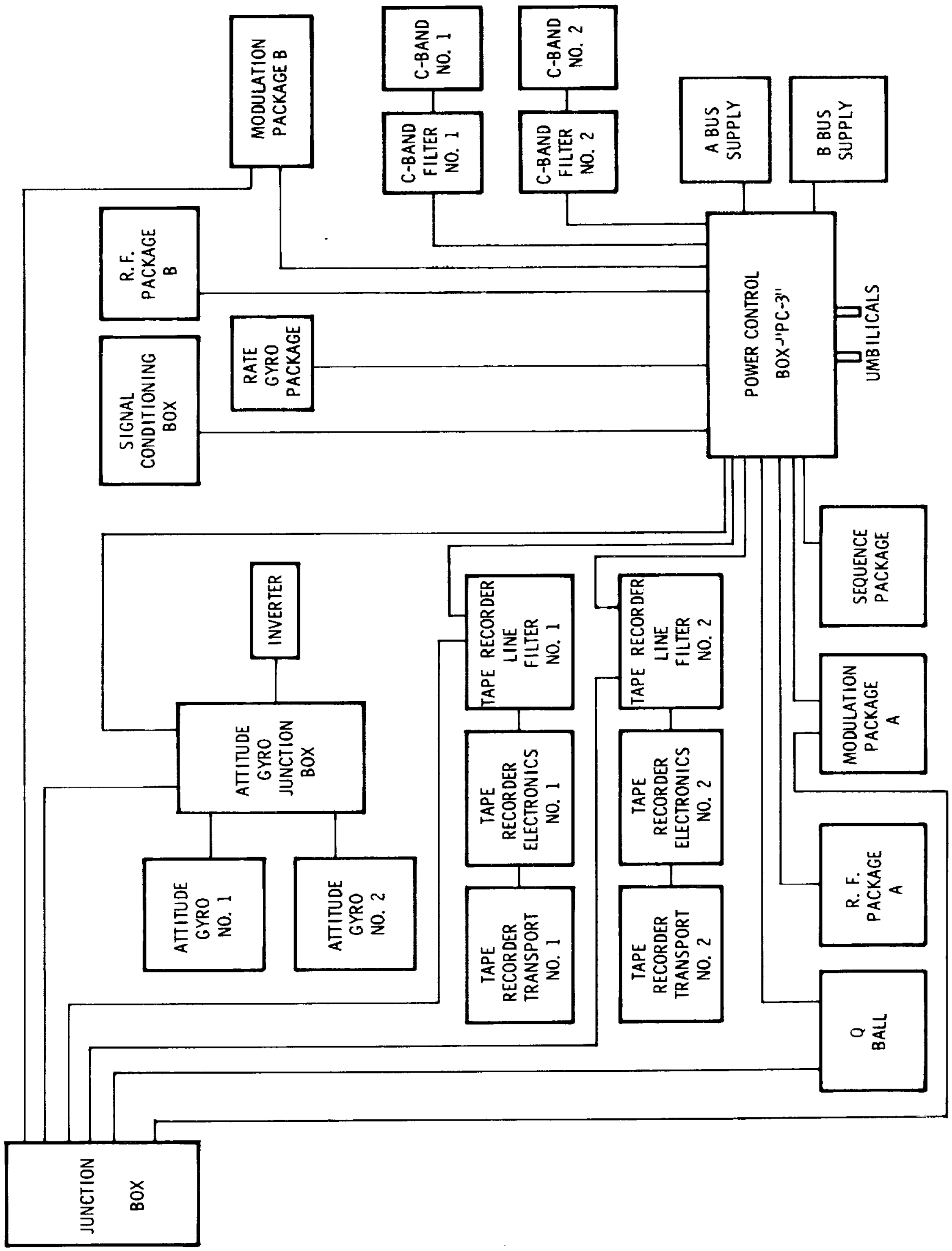

Figure 4-8. - Electrical power subsystem

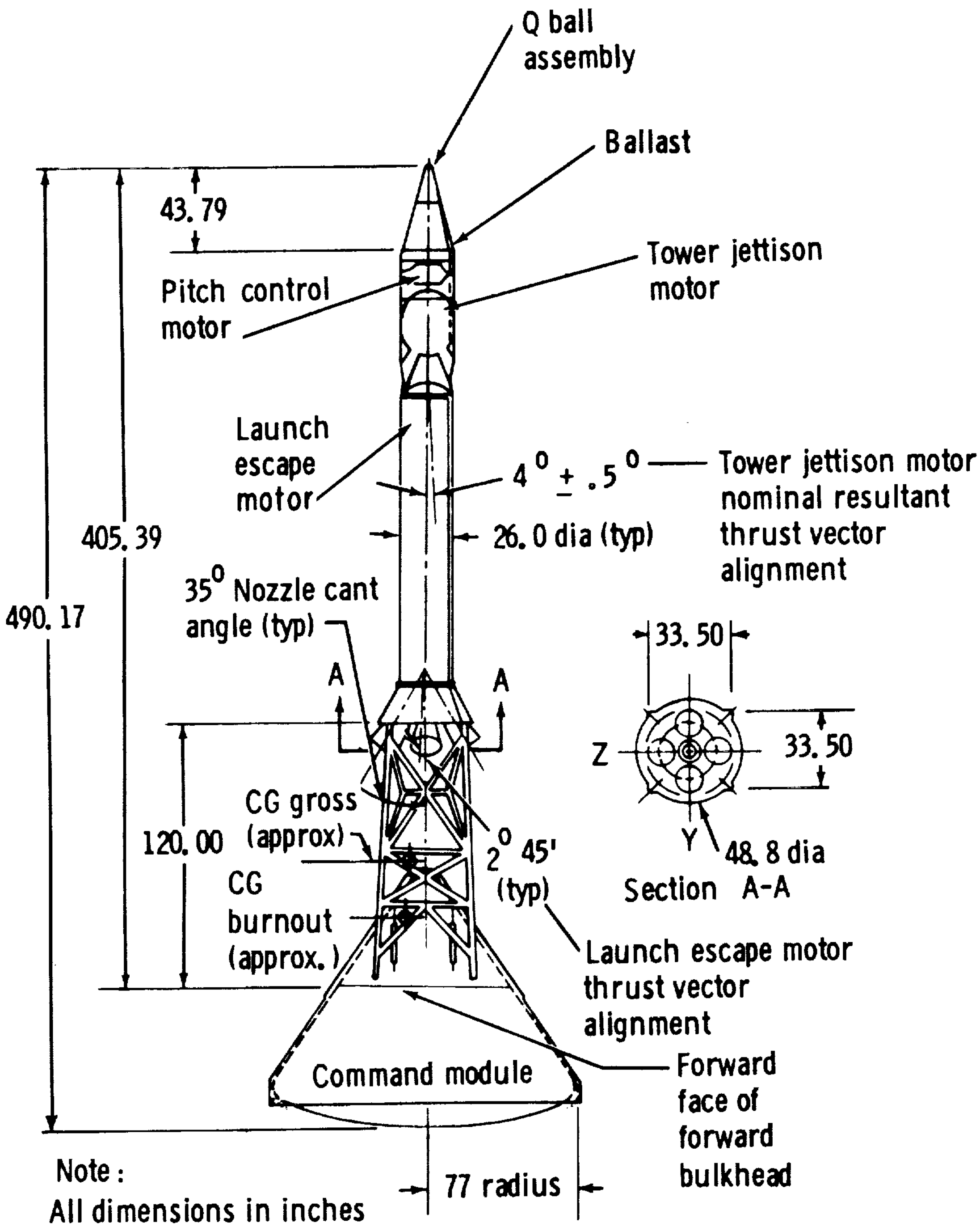

Figure 4-9. - Alinement and location of spacecraft rocket motors

[redacted]

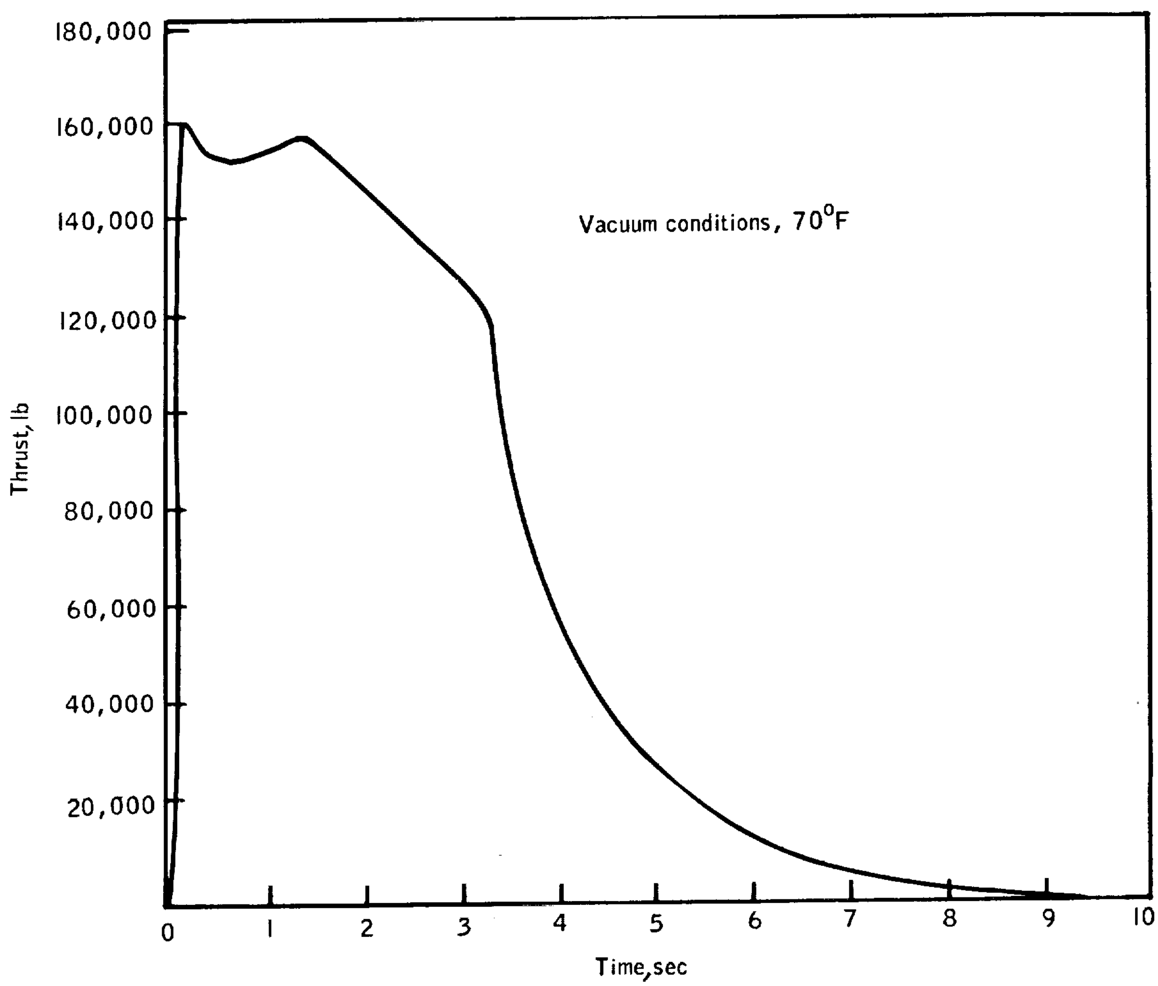

Figure 4-10. - Launch escape motor nominal thrust

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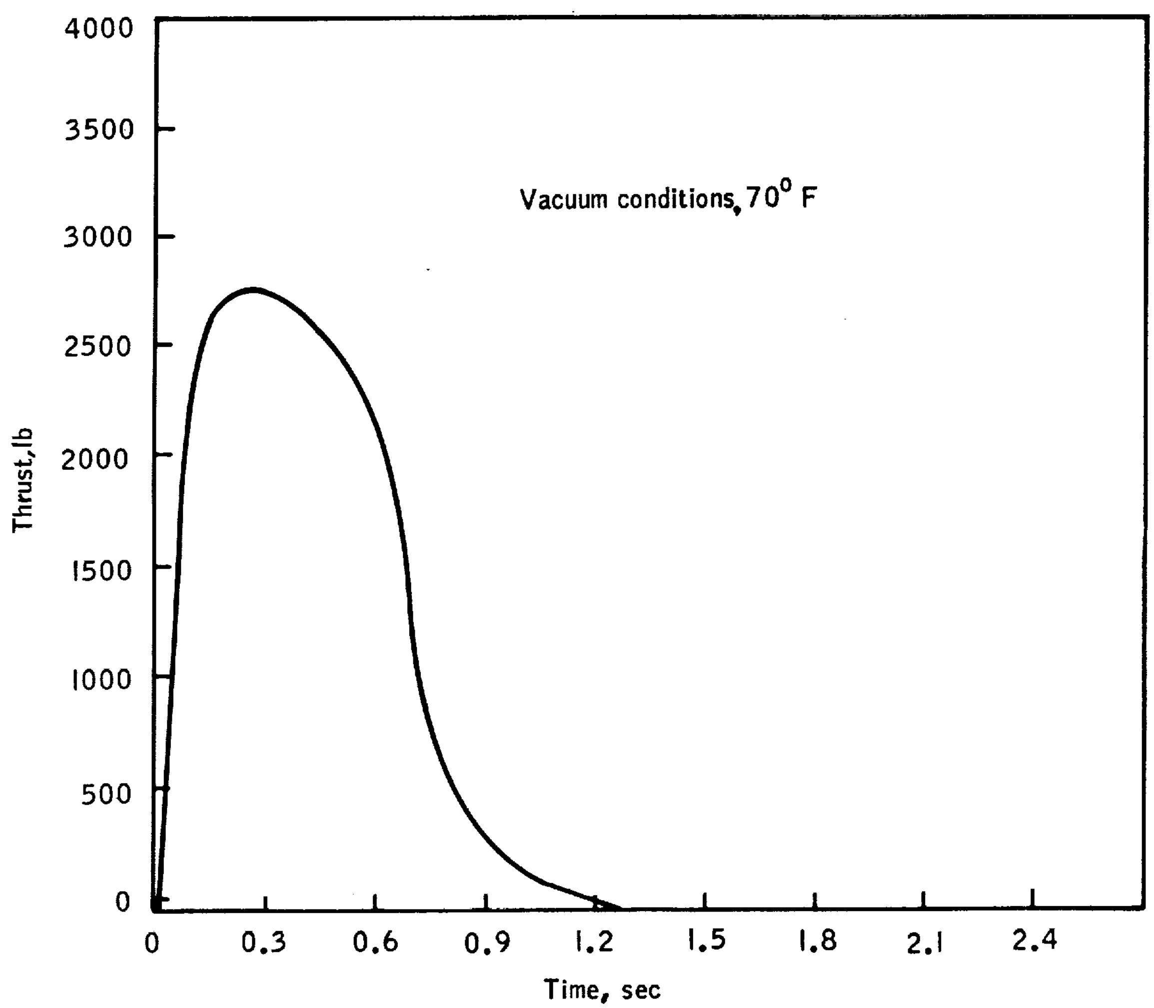

Figure 4-11. - Pitch control motor nominal thrust

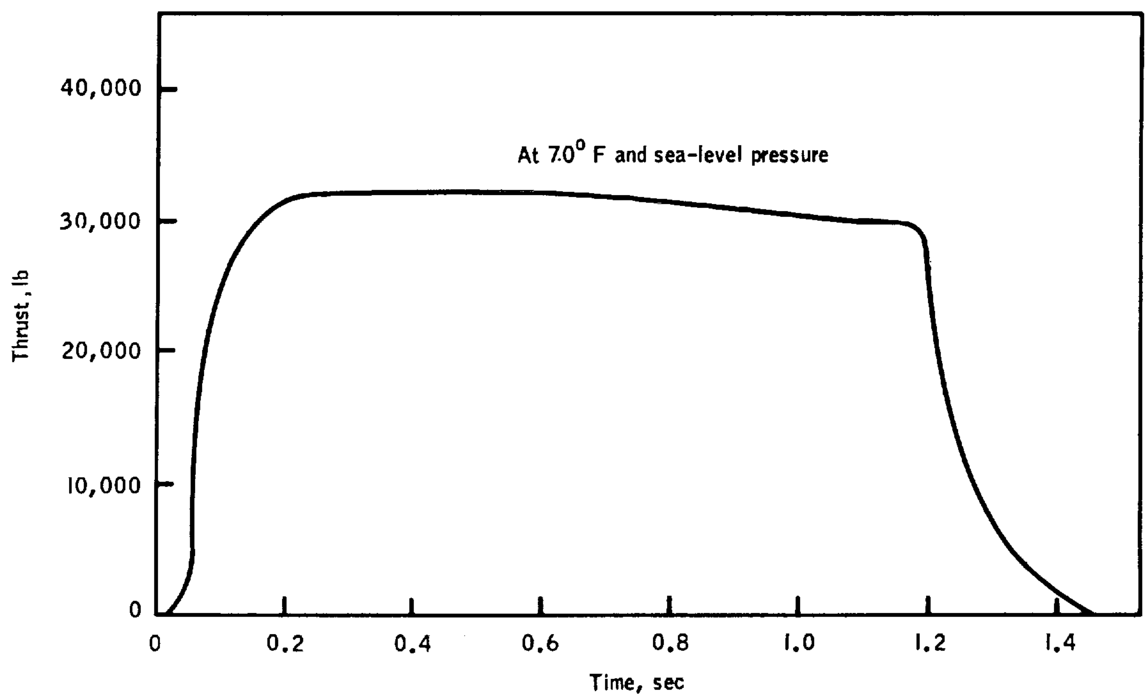

Figure 4-12. - Tower jettison motor nominal thrust

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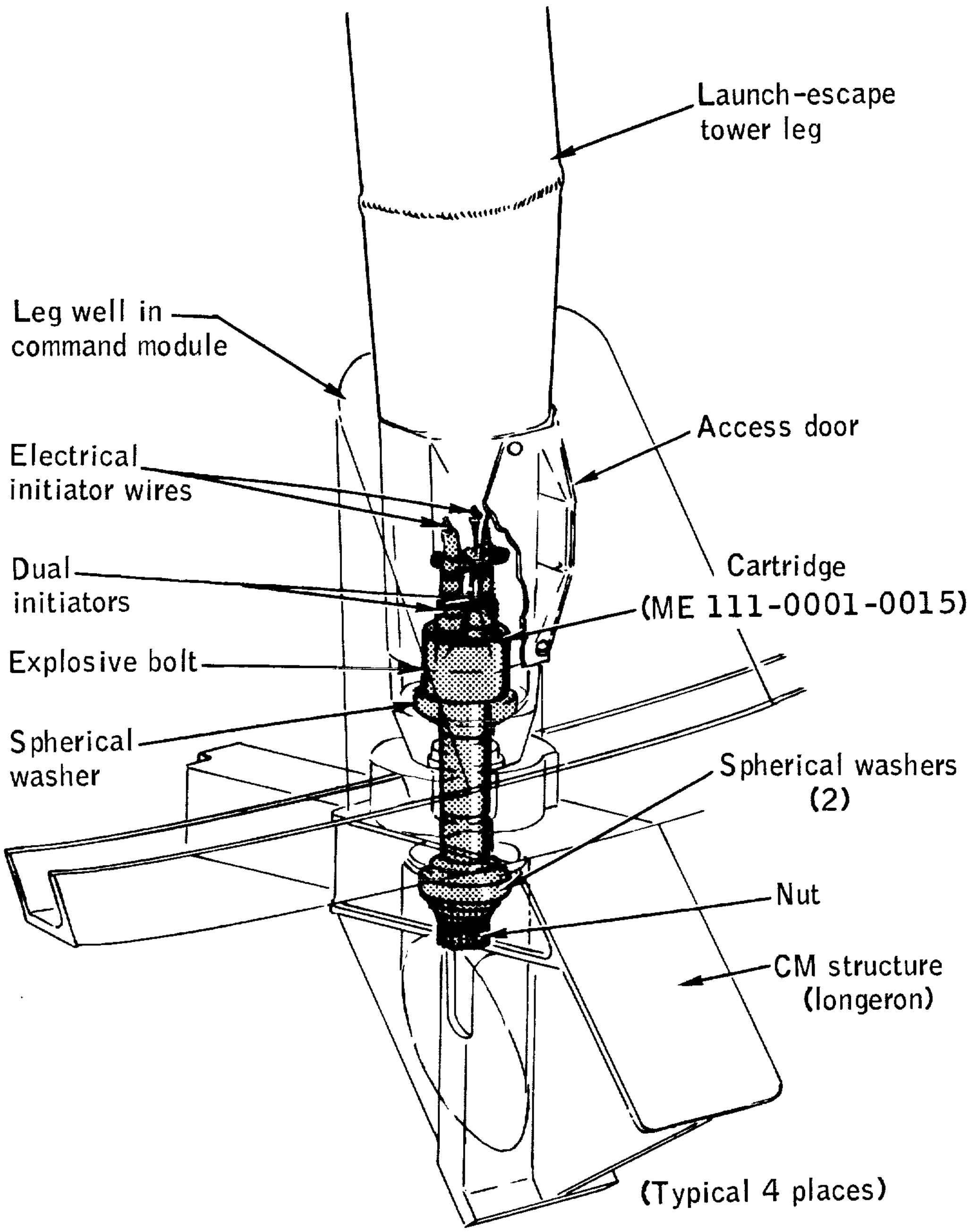

Figure 4-13. - Single mode explosive bolt installation

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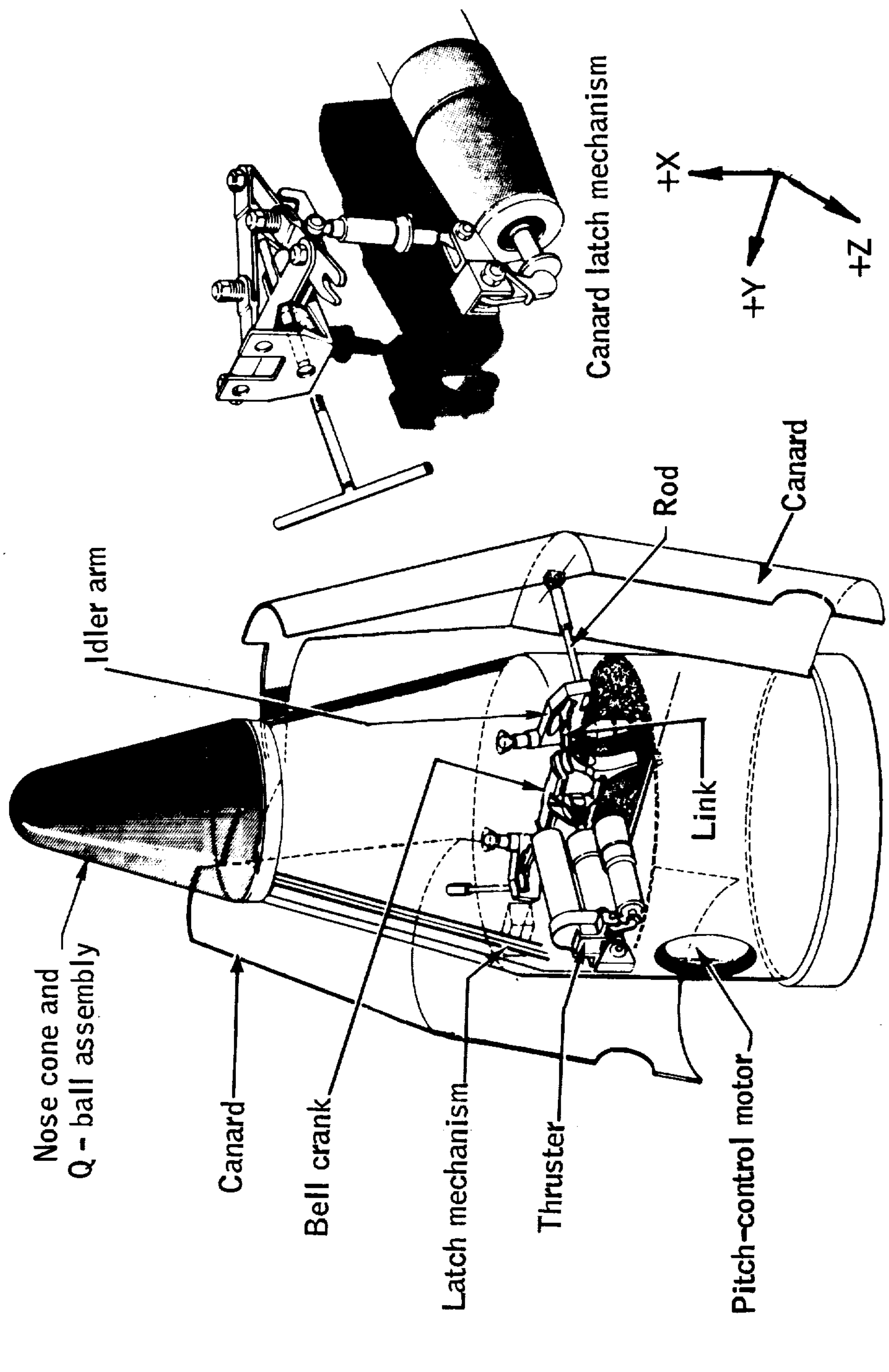

Figure 4-14. - LES canard subsystem

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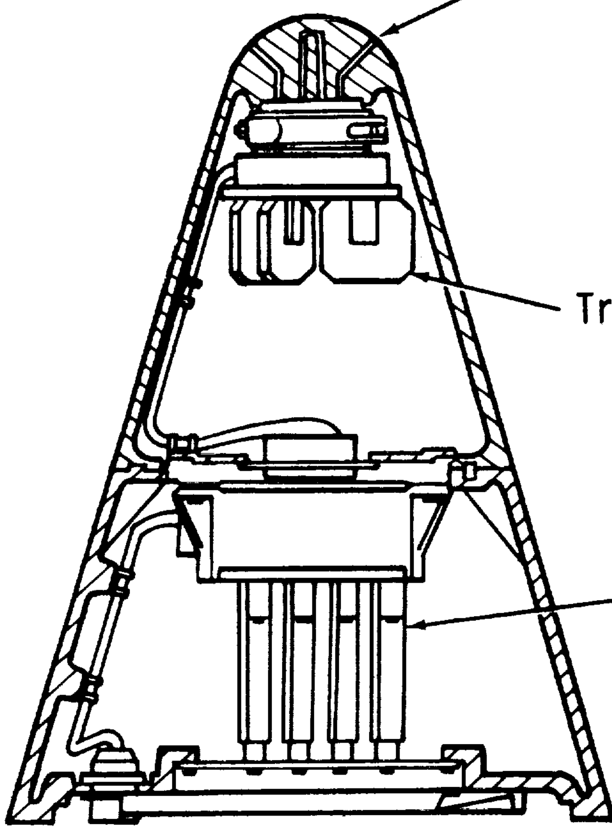

Figure 4-15. - Q-ball assembly

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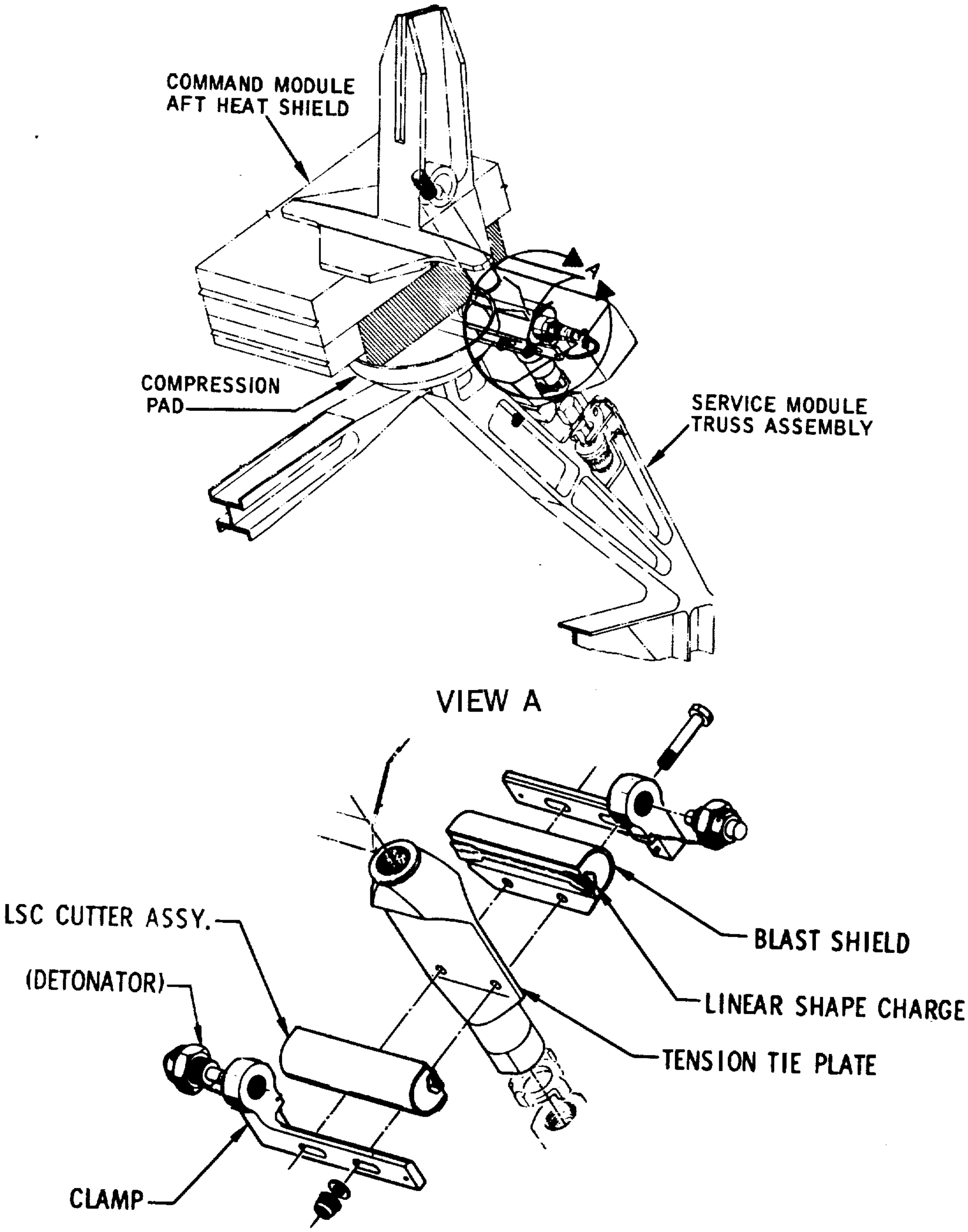

Figure 4-16. - CM-to-SM separation subsystem

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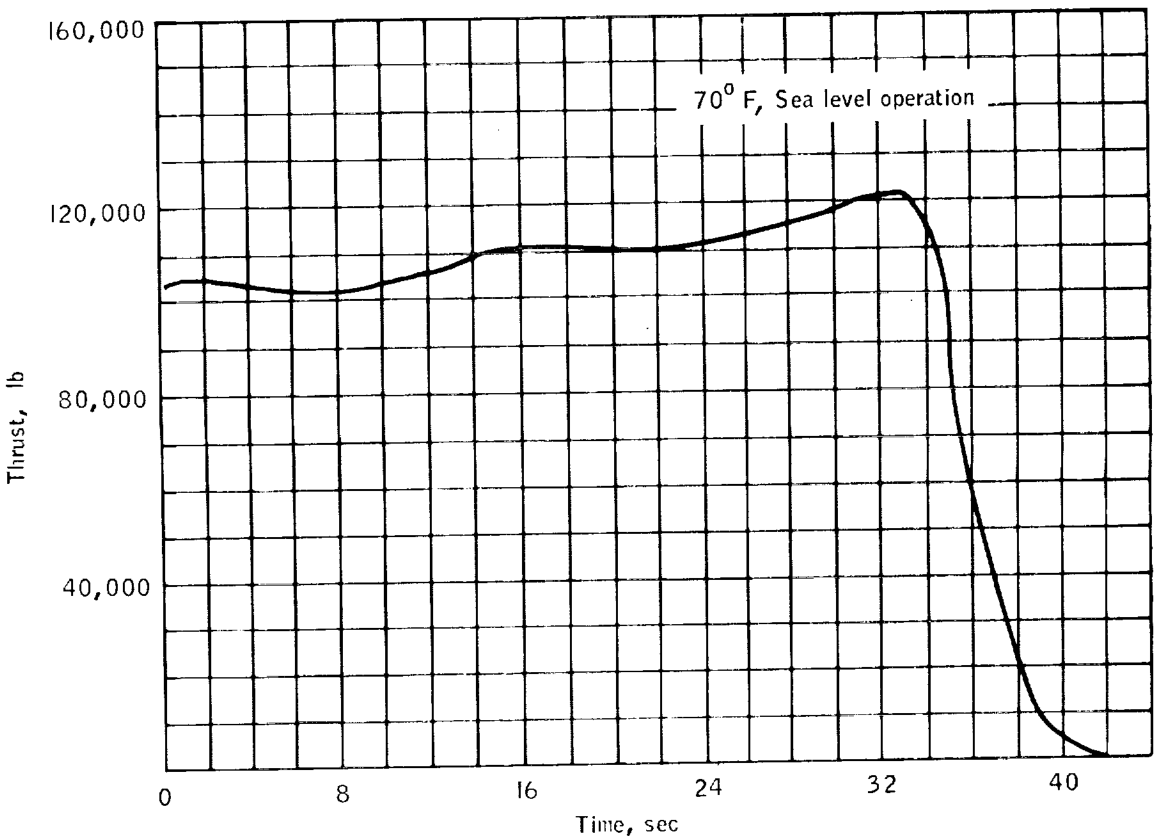

Figure 4-17. - Algol rocket motor nominal thrust

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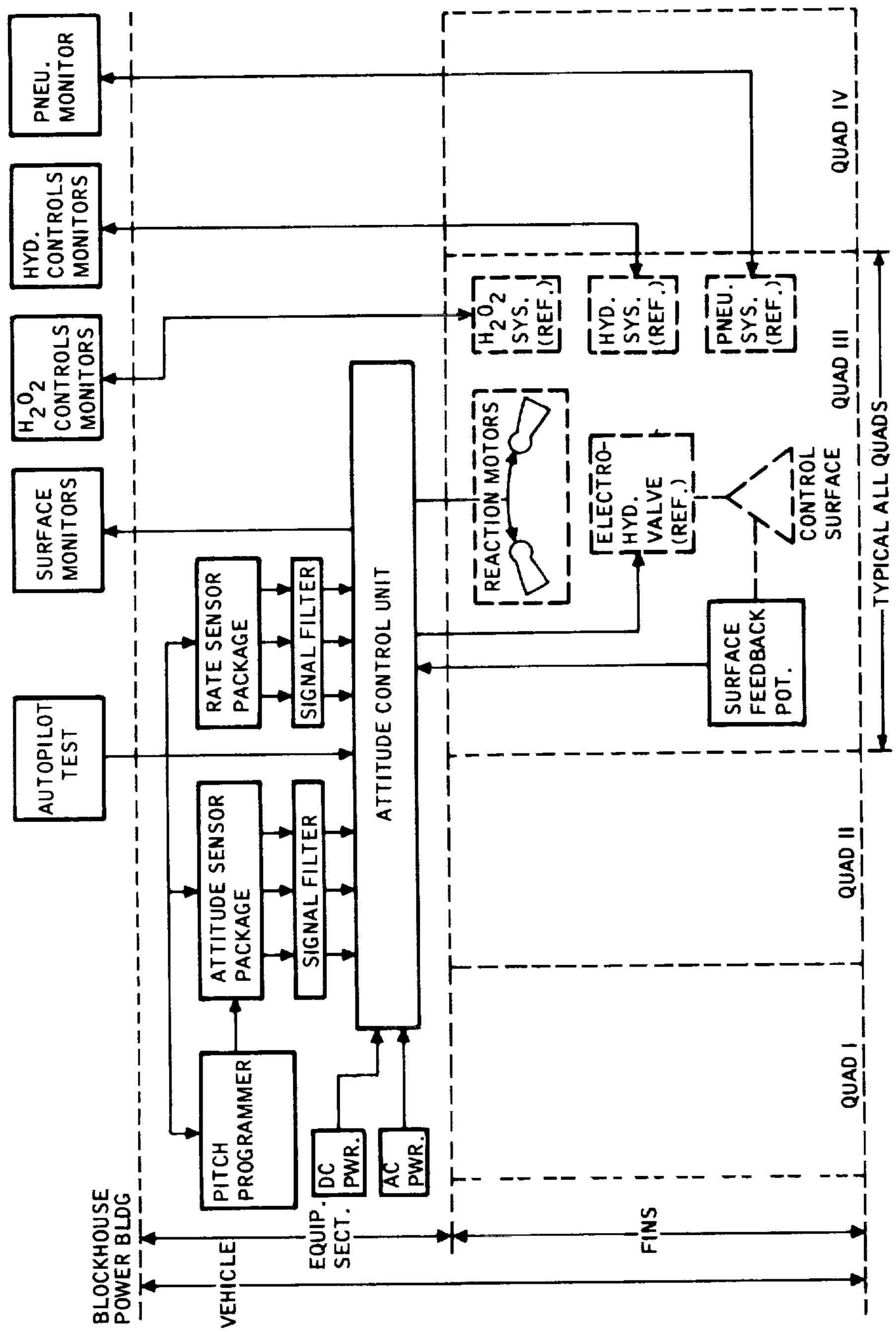

Figure 4-18. - Little Joe II attitude control subsystem

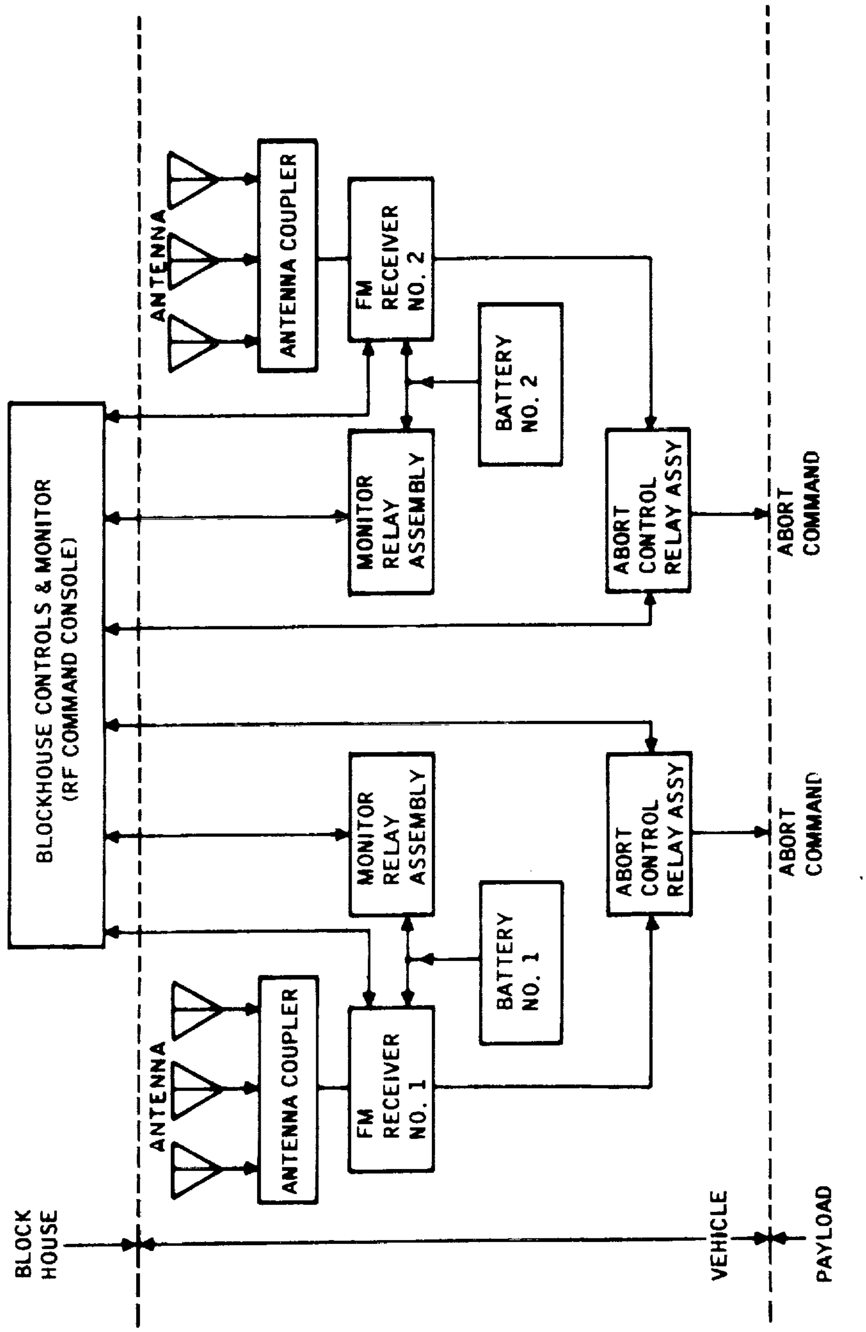

Figure 4-19. - Launch vehicle RF command subsystem block diagram

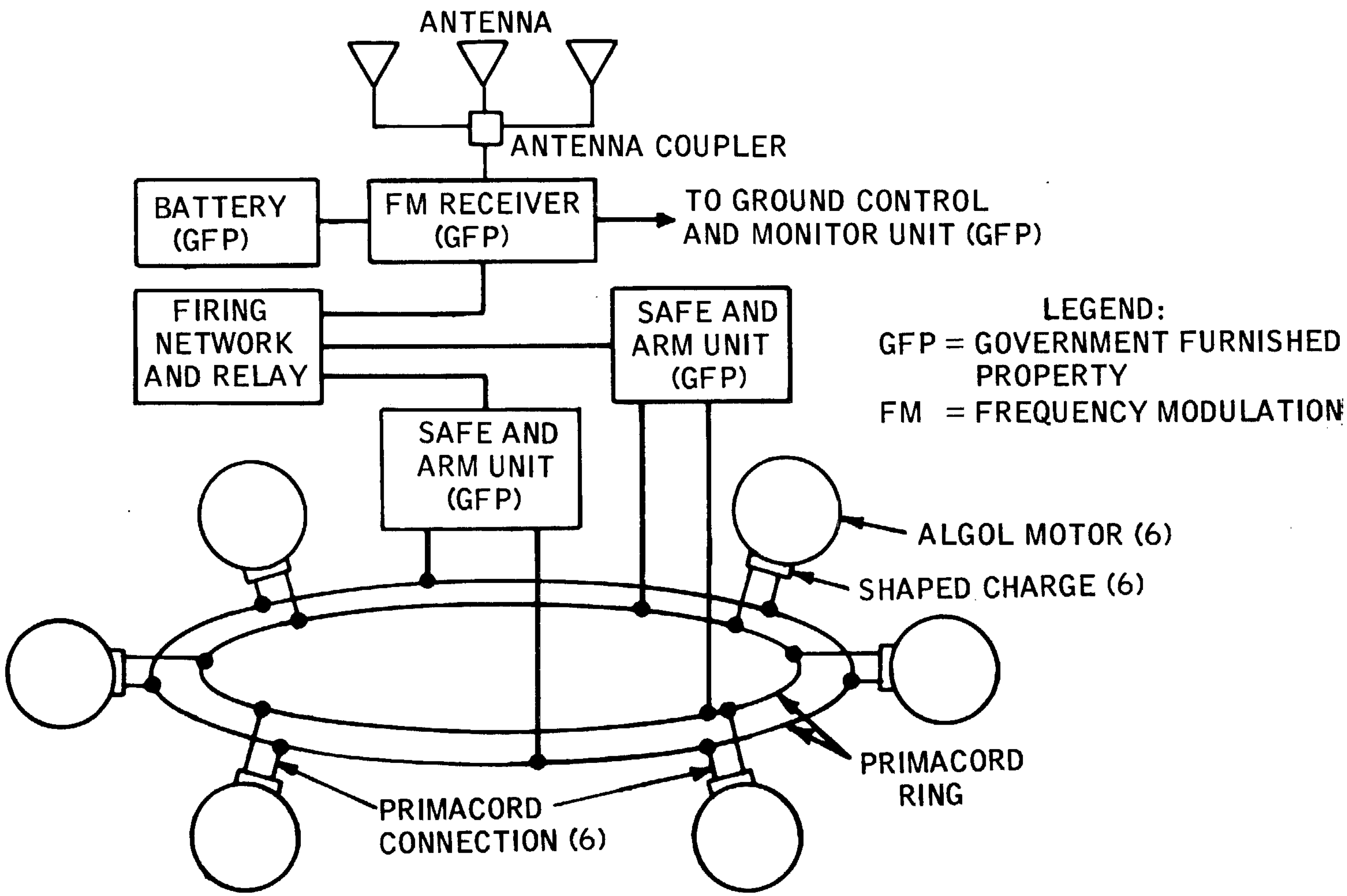

Figure 4-20. - Block diagram of destruct subsystem

5.0 TEST VEHICLE CONSTRAINTS

This section defines the minimum qualification testing that must be accomplished prior to launch. Failure to accomplish any of these tests will delay the flight. The constraints that apply to the Mission A-003 test vehicle follow:

Constraining Item	Constraint
1. Apex heat shield thrusters	Complete ground system and individual cartridge tests
2. Earth landing system	Completion of BP 19 drop tests at El Centro
(a) Drogue parachute disconnect	Delivery schedule
3. Mission sequencer	1. Satisfactory completion of minimum airworthiness tests 2. Add circuitry to delay arming of pyrotechnic bus until after heat shield jettison
4. Instrumentation	
(a) Tape recorders	Satisfactory completion of qualification tests <u>after</u> modifications at WSMR
(b) Signal conditioner	Completion of outstanding Engineering Orders
(c) Cameras	Delivery and qualification
5. LES Tower	Determine coefficient of stiffness - must be delivered to NASA structures group 4 weeks prior to launch
6. Backup abort timer relay Autronics 1890-1-85	Successful completion of environmental tests (vibration and voltage) on 3 time delay relays per STP 695-731-265-1
7. Final weight and balance report	Final report delivered to NASA structures group 4 weeks prior to launch

Constraining Item	Constraint
8. Boost protective cover	Satisfactory fit - rework per EO M273903 and EO M29269
9. Range safety system	Completion of tests
10. Reefing cutters used on both drogue parachutes and main parachutes	Completion of ground tests

6.0 INSTRUMENTATION REQUIREMENTS

6.1 Spacecraft Data Acquisition Subsystem

The spacecraft data acquisition subsystem (figure 6-1) is described in detail in reference 10. A detailed list of parameters measured is available in reference 4; however, the measurements are summarized in table 6-I.

6.1.1 Onboard tape recorders.- The two onboard tape recorders, used to record the telemetry outputs and some measurements requiring high frequency response that are not telemetered, are the primary data recording devices. Each tape recorder consists of a tape transport, recorder electronics, and a remote control box. The tape is 1 inch wide and includes 14 tracks. Operating at 15 inches per second, the tape transport includes sufficient tape for approximately 30 minutes of recording.

6.1.2 Telemetry subsystem.- The telemetry subsystem consists of two PAM/FM/FM telemetry sets, each including 17 measurement channels. Continuous measurements are carried on 14 channels of link A; the output of a 90 x 10 commutator and a 90 x 1.25 commutator are carried on two of these channels. Link B carries continuous measurements on 13 channels; one of which carries the output of a 90 x 10 commutator.

Each telemetry set consists of transducer signal conditioning equipment, commutators, voltage controlled oscillators (VCO's) and a modulating transmitter. The transmitters of links A and B operate at 237.8 mc and 247.3 mc, respectively.

The two RF signals are combined by a multiplexer and the composite signal is equally distributed to four slot antennas by three power dividers. The four slot antennas, located below the separation line of the forward heat shield and spaced every 90 degrees around the command module, transmit the signal to ground stations.

Calibration of the TM subsystem will be accomplished prior to launch and will consist of R and Z and 5-point calibration. The R and Z calibration will consist of 15 and 85 percent of full scale signals. The 5-point calibrations will consist of 0, 1.25, 2.5, 3.75, and 5 volts or equivalent signal percentages. During the 5-point calibration, the commutators on IRIG channel E and the 90 x 1.25 channel will be switched out of the circuit. There will be no inflight calibration. The timing, accuracy, and format is compatible with the type B code system of the Time Format Standards, reference 10.

6.1.3 End instruments and signal conditioning.- The instrumentation transducers are devices such as thermocouples, vibration sensors, rate and attitude gyros, break-wire systems, accelerometers, pressure transducers, etc., that measure the desired quantities. Signals from these transducers are converted to a standard measuring scale (signal conditioned) before they are applied to the commutators or VCO's. The signal conditioning equipment includes bridge adjustment units, thermocouple compensation networks, frequency converters, phase sensitive demodulators, and similar devices.

6.1.4 C-band transponders.- Two C-band transponders are installed to aid in tracking the spacecraft. Each transponder operates independently and drives two slot antennas through a power divider. The four cavity-backed helix antennas are spaced 90 degrees apart around the CM, below the separation line of the forward heat shield.

6.1.5 Camera installations.- Three cameras are installed in the spacecraft. Camera locations are shown in figure 6-2 and camera operating times are shown in figure 6-3. Each camera subsystem is an independent unit, including a power supply, a timer to turn the camera on, a cut-off switch to stop the camera, and a tri-pulse timing generator.

Camera No. 1, installed in the egress hatch, operates for approximately 110 seconds at 200 frames per second (fps). The primary objective of this camera is to record the following events which occur forward of the CM:

(a) Boost protective cover separation

(b) Dual drogue parachute deployment and dereefing

(c) Drogue parachute release

(d) Main parachute extraction, deployment, and dereefing

Camera No. 2, installed in the service module, operates for approximately 40 seconds at 400 fps. The function of this camera is to record the separation of the command module from the service module.

Camera No. 3, installed in the launch escape tower, operates for approximately 85 seconds at 32 fps. It records the shape of the launch escape motor plume and performance of the boost protective cover. Since this camera is activated at T+15 seconds, the effects of the critical part of the boost phase on the boost protective cover are recorded.

6.1.6 Thermal paint samples.- Temperature sensitive paint will be applied to the exterior of the boost protective cover and the surface of the parachute compartment.

6.1.7 Glass samples.- Four glass samples, approximately 3" x 5", will be attached to the command module in locations approximating the rendezvous and side windows on Block I spacecraft.

6.2 Launch Vehicle Data Acquisition Subsystem

Launch vehicle data acquisition is described in detail in reference 9. A detailed list of quantities measured is available in reference 4; however, the measurements are summarized in table 6-I.

The signal outputs for all measured quantities on the launch vehicle are routed through LV-to-SM-to-CM umbilical cables to the command module telemetry subsystem.

6.3 Ground Measured Data Acquisition Subsystem

In addition to the inflight measurements, data are also sent by landline from the vehicle to the ground telemetry trailer. The data sent via landline are:

(a) LES motor star point temperature

(b) LES motor star valley temperature

(c) LES motor case exterior surface temperature (3 places)

TABLE 6-I.- SUMMARY OF INSTRUMENTATION FOR FLIGHT OF BP-22

SPACECRAFT, APOLLO MISSION A-003

Quantity	Measurement description	Commutated (COM) or continuous (CT) (if applicable)
	Pressures:	
34	CM conical surface	COM
4	CM base	COM
1	CM interior	COM
9	SM fluctuating	CT
1	CM LES plume impingement	COM
1	LES engine base	COM
1	Launch escape motor chamber	CT
1	Pitch control motor chamber	CT
1	Command module interior	COM
1	Dynamic pressure	COM
	Temperatures:	
13	CM calorimeter body	COM
3	Tower calorimeter body	COM
1	CM interior	COM
3	Tower leg	COM
2	Telemetry RF amplifier	COM
1	Pitch control motor case	COM
1	Tower jettison motor case	COM
1	LES motor case	COM
	Calorimeters:	
13	CM heat flux	COM
3	Tower heat flux	COM
	Accelerations, rates, vibrations, and strains:	
4	CM axial accelerations	CT
2	Tower axial accelerations	CT
3	Rate gyro outputs	CT

TABLE 6-I.- SUMMARY OF INSTRUMENTATION FOR FLIGHT OF BP-22

SPACECRAFT, APOLLO MISSION A-003 - Continued

Quantity	Measurement description	Commutated (COM) or continuous (CT) (if applicable)
4	Attitude gyro outputs (2 roll)	COM
2	CM axial vibrations	CT
2	CM radial vibrations	CT
30	SM vibrations	CT
2	Canard actuator link strains	CT
	Electrical functions:	
2	DC voltage, main bus	COM
2	DC voltage, logic bus	COM
2	DC voltage, LES pyro bus	COM
1	DC current, total (instrumentation)	COM
2	Differential PDM (pulse duration modulation) (90 X 10 comm)	CT
2	Mixer output	CT
7	PDM reference voltages	COM
6	PDM synchronization pulses	COM
	Events:	
1	Onboard timer	-
2	Gyro segment switch	COM
2	GSE abort enable system	COM
4	Abort initiate relay closure	COM
2	Backup abort timer closure	COM
2	LES sequencer start signal	COM
2	LES/pitch control motor fire relay closure	COM
2	CM/SM separation relay closure	COM
1	Physical monitor CM/SM separation	COM
4	Canard deploy relay closure	COM
1	Canard actuator displacement	CT
2	Tower jettison and separation relay closure	COM
4	ELS sequencer start relay closure	COM
2	Baroswitch lock-in relay closure	COM

TABLE 6-I.- SUMMARY OF INSTRUMENTATION FOR FLIGHT OF BP-22

SPACECRAFT, APOLLO MISSION A-003 - Concluded

Quantity	Measurement description	Commutated (COM) or continuous (CT) (if applicable)
2	Drogue deploy relay closure	COM
2	Physical monitor drogue chute	COM
2	Main chute deploy - drogue release relay closure	COM
2	CM forward heat shield jettison	COM
	Other:	
1	Angle of attack	COM
1	Angle of sideslip	COM
3	Cameras	-
	Little Joe II:	
6	Algol motor chamber pressures	COM
2	Fin actuator hydraulic pressure	CT
1	Pitch rate	CT
1	Roll rate	CT
1	Yaw rate	CT
2	Fin control surface position	CT
153	Total commutated	
68	Total continuous	
4	Other	

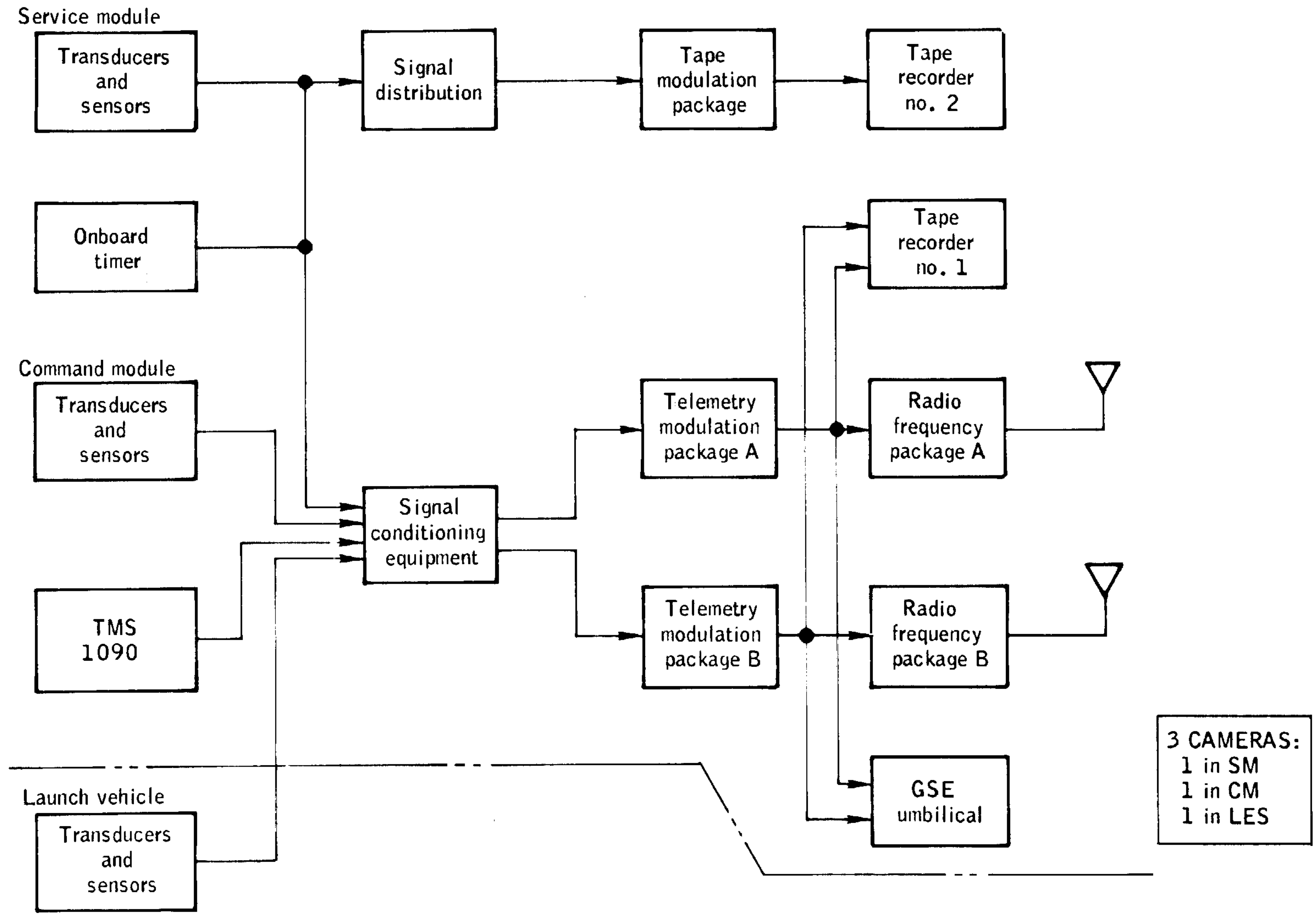

Figure 6-1. - BP-22 spacecraft instrumentation block diagram

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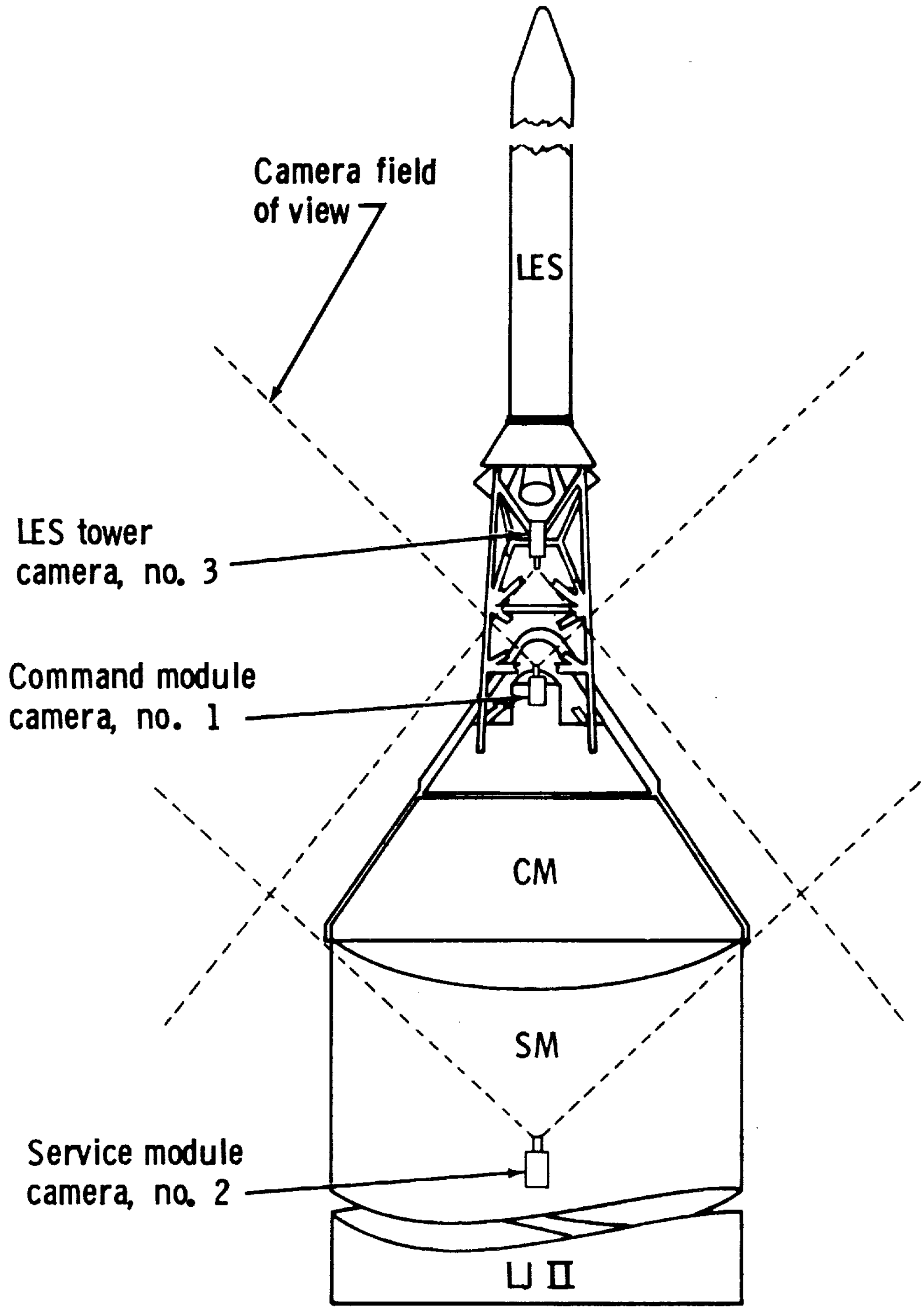

Figure 6-2. - BP-22 spacecraft camera locations

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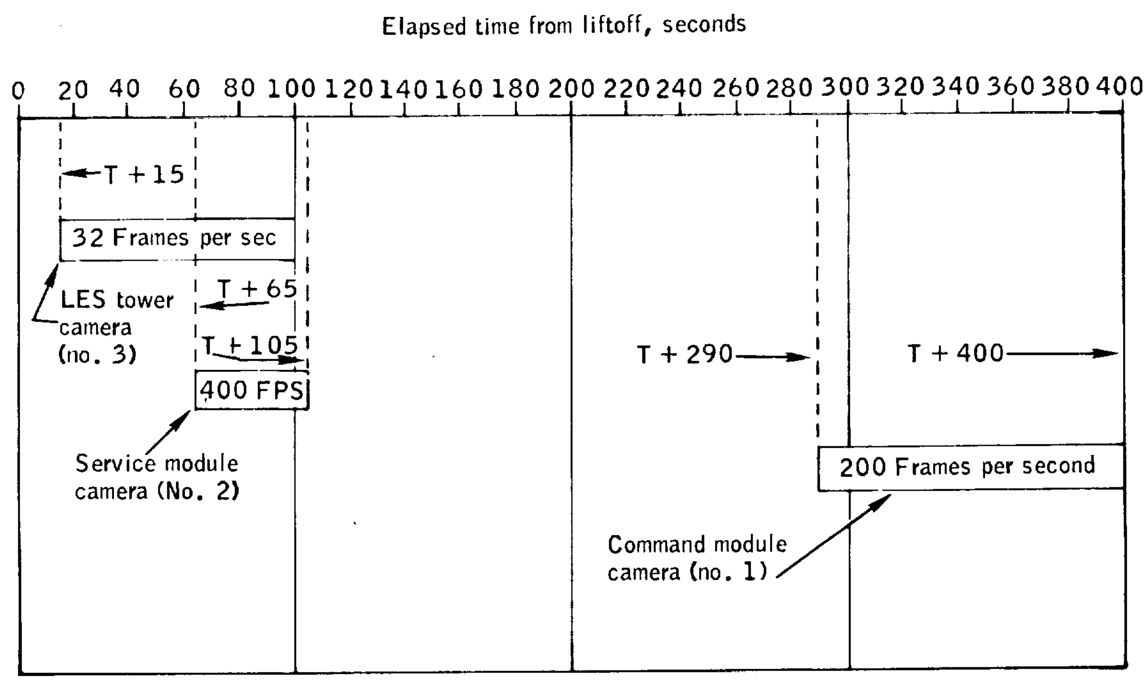


Figure 6-3. - Mission A-003 spacecraft camera operating characteristics

7.0 TRACKING AND SUPPORT DATA REQUIREMENTS

7.1 General

The general tracking and support data requirements are specified in the Program Requirements Document. Detailed data requirements such as sampling rates, accuracies, optical coverage, and referencing of data are covered in Operations Requirements Document 90202 (reference 3). The following paragraphs include a brief summary of the broad requirements.

7.2 Radar Tracking Data

The spacecraft will be tracked by radar from launch to impact and azimuth, elevation, and slant range data will be obtained. Two C-band transponders are installed in the command module for tracking purposes.

7.3 Phototheodolite and Engineering Photographic Data

7.3.1 Phototheodolite data.- Phototheodolite tracking is required to provide vehicle position acceleration, velocity, and attitude data. These data are required during certain portions of the flight.

7.3.2 Engineering photographic data.- Engineering photographic data has timing marks for correlation with flight time and is used for data analysis. Fixed high-speed, 35 mm color cine cameras will be located in a circle around the launch pad to record the initial launch conditions. In addition, a sufficient number of 35 mm, color and black and white telescopic tracking cameras will be assigned to cover significant phases of the flight from launch to impact.

7.4 Documentary Photographic Films

The transportation, preflight preparations, checkout, flight, and postflight activity of BP-22 spacecraft will be recorded by motion picture and still photography for historical purposes.

7.5 Meteorological Data

The White Sands Missile Range will be required to furnish weather data, consisting of wind direction and velocity, air density, temperature, and relative humidity, as a function of altitude.

7.6 Real-Time Data Display Systems

The real-time data display systems will be used to provide real-time monitoring of key flight parameters.

7.6.1 Real-time data display system for vehicle flight.- The real-time data display system (RTDS), used to present instantaneous test vehicle performance data, utilizes three FPS-16 (C-Band) radars feeding signals through a Kineplex link to an IBM 7094 computer in the display room of Building 1512 at WSMR. In the display room there are four plotting boards, one digital display panel, and one WSMR representative console. The plotboards will consist of:

(a) Plotboard A - Flight path angle as a function of altitude

(b) Plotboard B - Altitude as a function of vehicle velocity

(c) Plotboard C - Crossrange distance as a function of downrange distance and altitude as a function of downrange distance

(d) Nondesignated plotboard - Acceleration (from telemetry) as a function of time and acceleration (from radar) as a function of time

8.0 TEST MANAGEMENT ORGANIZATION AND PRELAUNCH OPERATIONS

8.1 Checkout at Contractor Facility

Checkout of the launch vehicle and the spacecraft prior to delivery to NASA is the responsibility of the respective contractors and will occur at the contractor's facility (factory) immediately prior to delivery to NASA. However, checkout will be monitored by NASA test and checkout teams. The typical functional relationship is shown in figure 8-1 for the spacecraft team.

8.2 Preflight Checkout and Launch at WSMR

Preflight checkout and launch operations at WSMR will be directed by NASA through use of the launch operations team organization shown in figure 8-2. This team is composed of many of the same members of the team shown in figure 8-1.

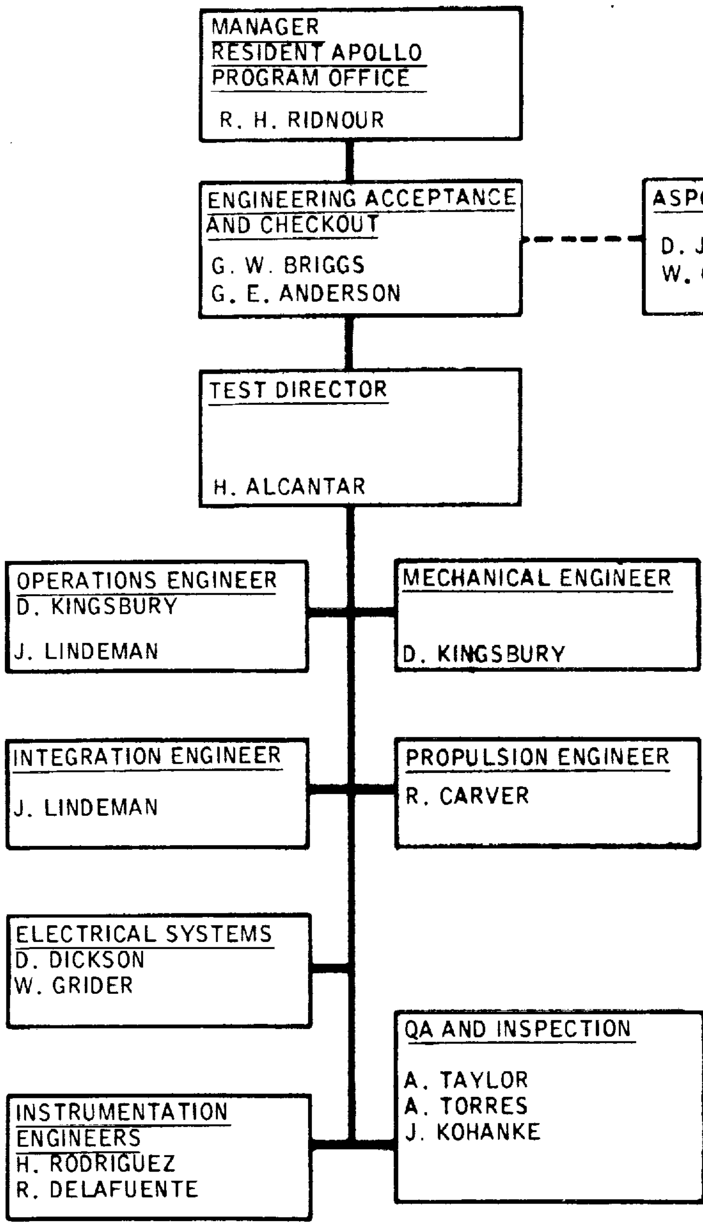
8.3 Prelaunch Operations

The schedules for spacecraft preflight preparation at the factory (NAA, Downey, California) and at the launch site (WSMR) are shown in figures 8-3 (a) and 8-3 (b), respectively.

The schedules for launch vehicle preflight preparation at the factory (General Dynamics-Convair, San Diego, California) and at the launch site (WSMR) are shown in figures 8-4 (a) and 8-4 (b), respectively.

An approximate schedule for precountdown and countdown of the spacecraft and launch vehicle are shown in figures 8-5 (a) and 8-5 (b), respectively.

The schedule figures in this section are for information only and are not to be used for test scheduling or planning.



NOTES:

1. OTHER WSMR SUPPORT WILL BE SECURED THROUGH H. R. VAN GOEY, WSMR, AS REQUIRED.
2. SUBSYSTEM MANAGER SUPPORT WILL BE SECURED FROM EACH SUBSYSTEM MANAGER AS REQUIRED.

Figure 8-1. - NASA checkout team for spacecraft

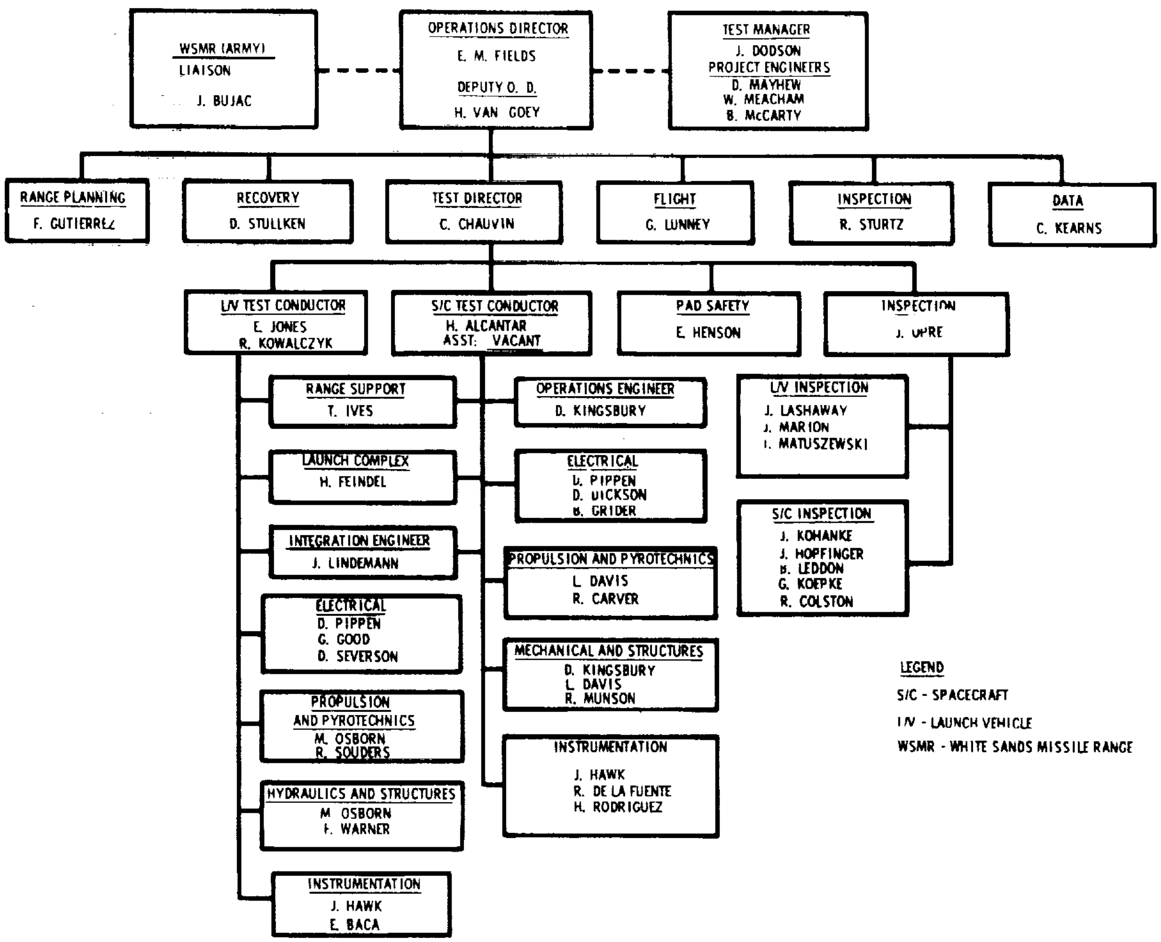


Figure 8-2. - NASA launch operations team

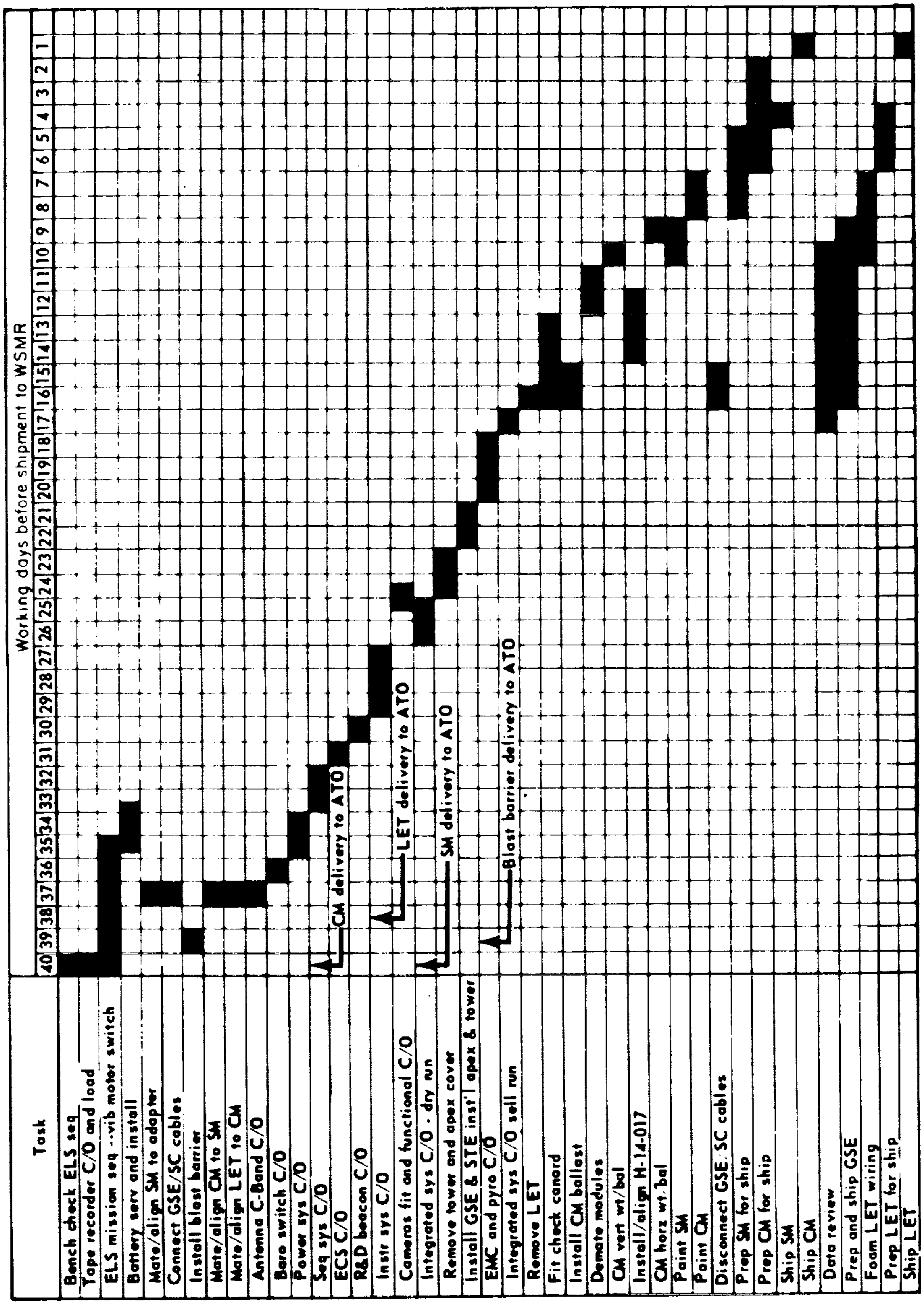


Figure 8-3. - Typical spacecraft test schedules (a) Downey test operations

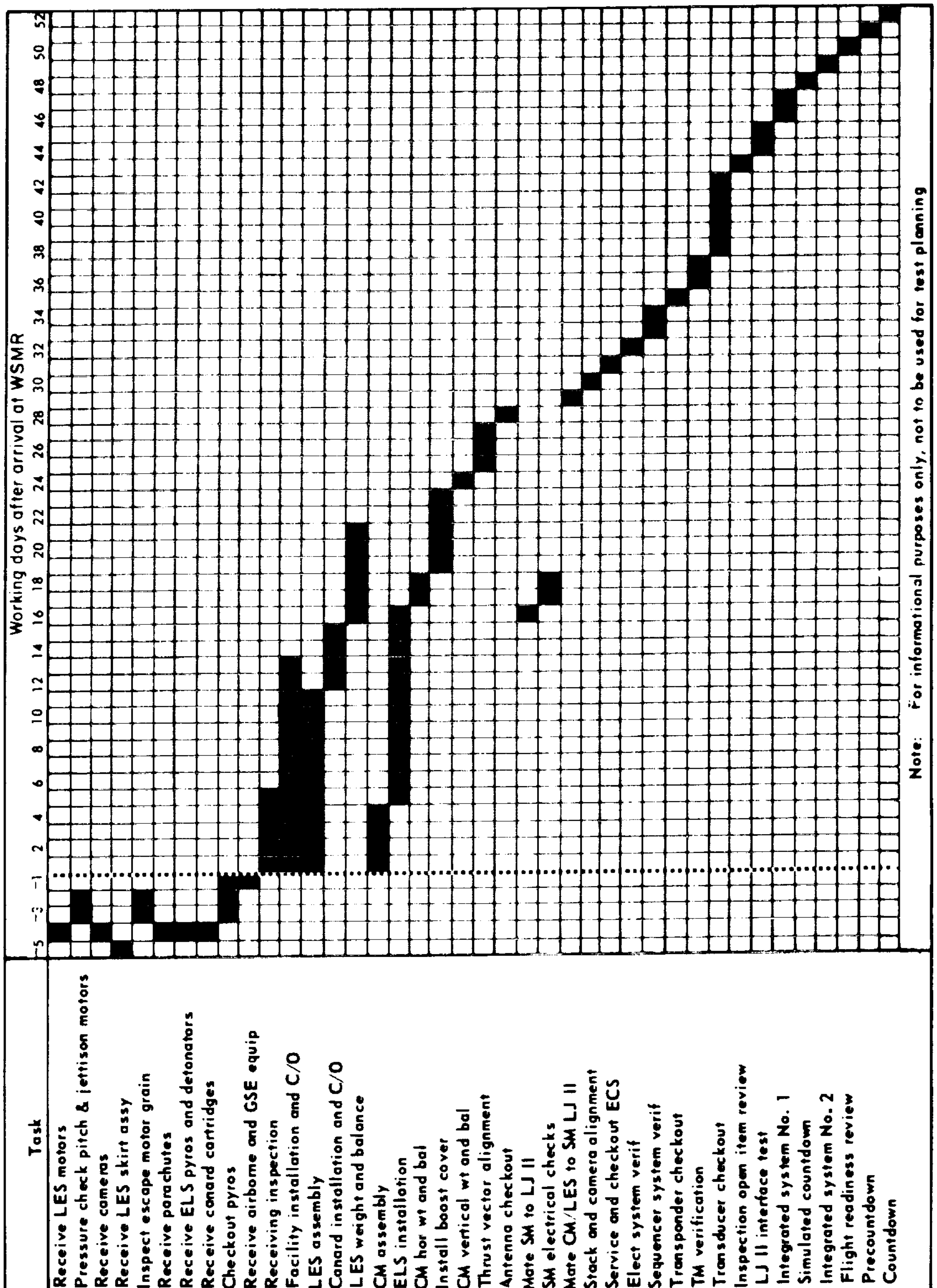


Figure 8-3. - Typical spacecraft test schedules (b) WSMR test operations

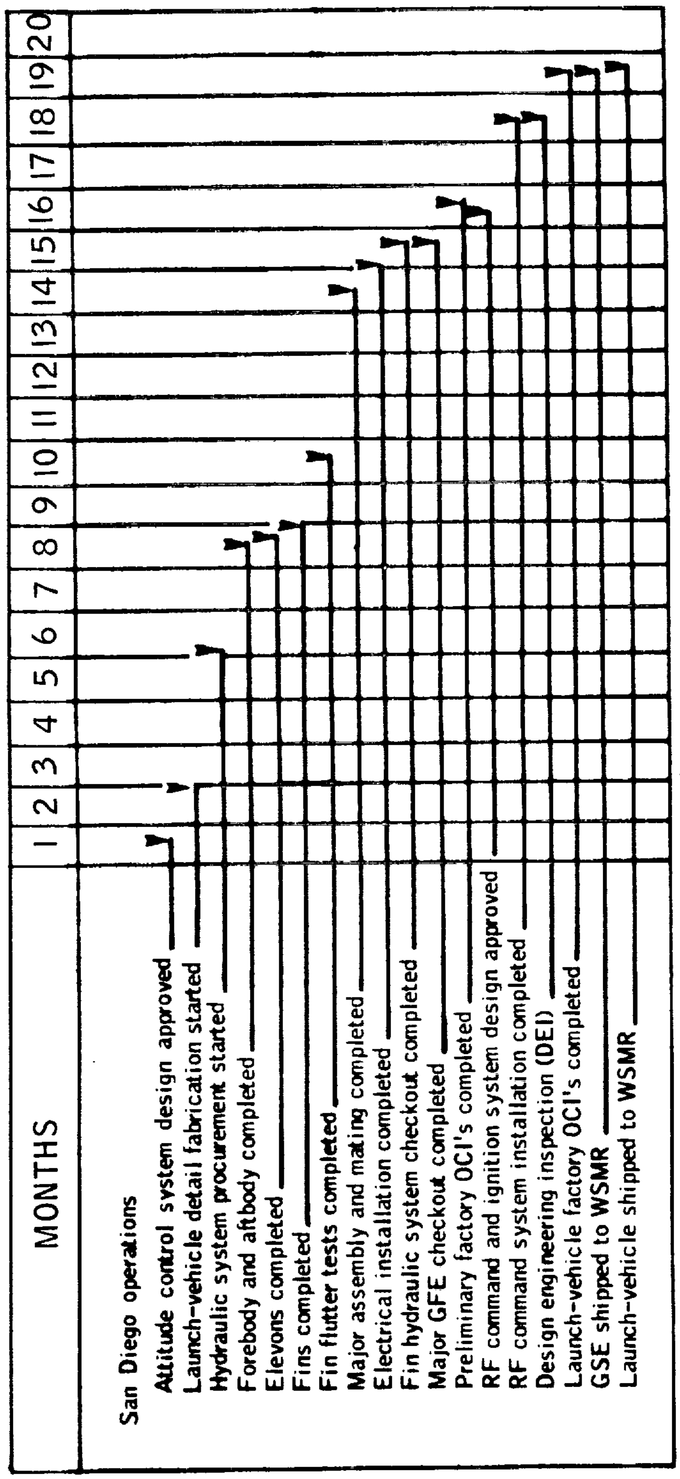


Figure 8-4. - Typical launch vehicle test schedules

(a) General Dynamics/Convair test operations

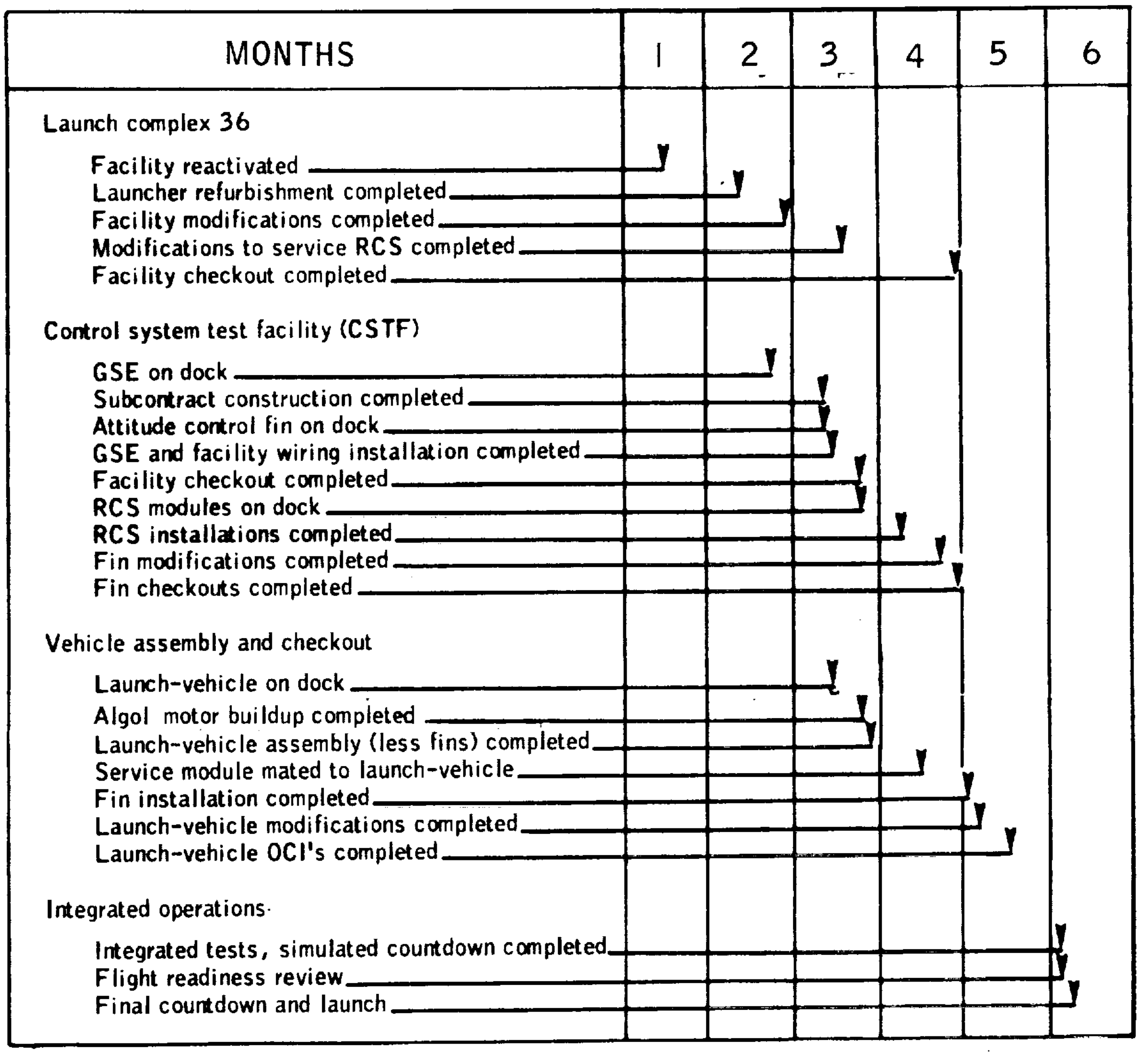


Figure 8-4. - Typical launch vehicle test schedules
(b) WSMR test operations

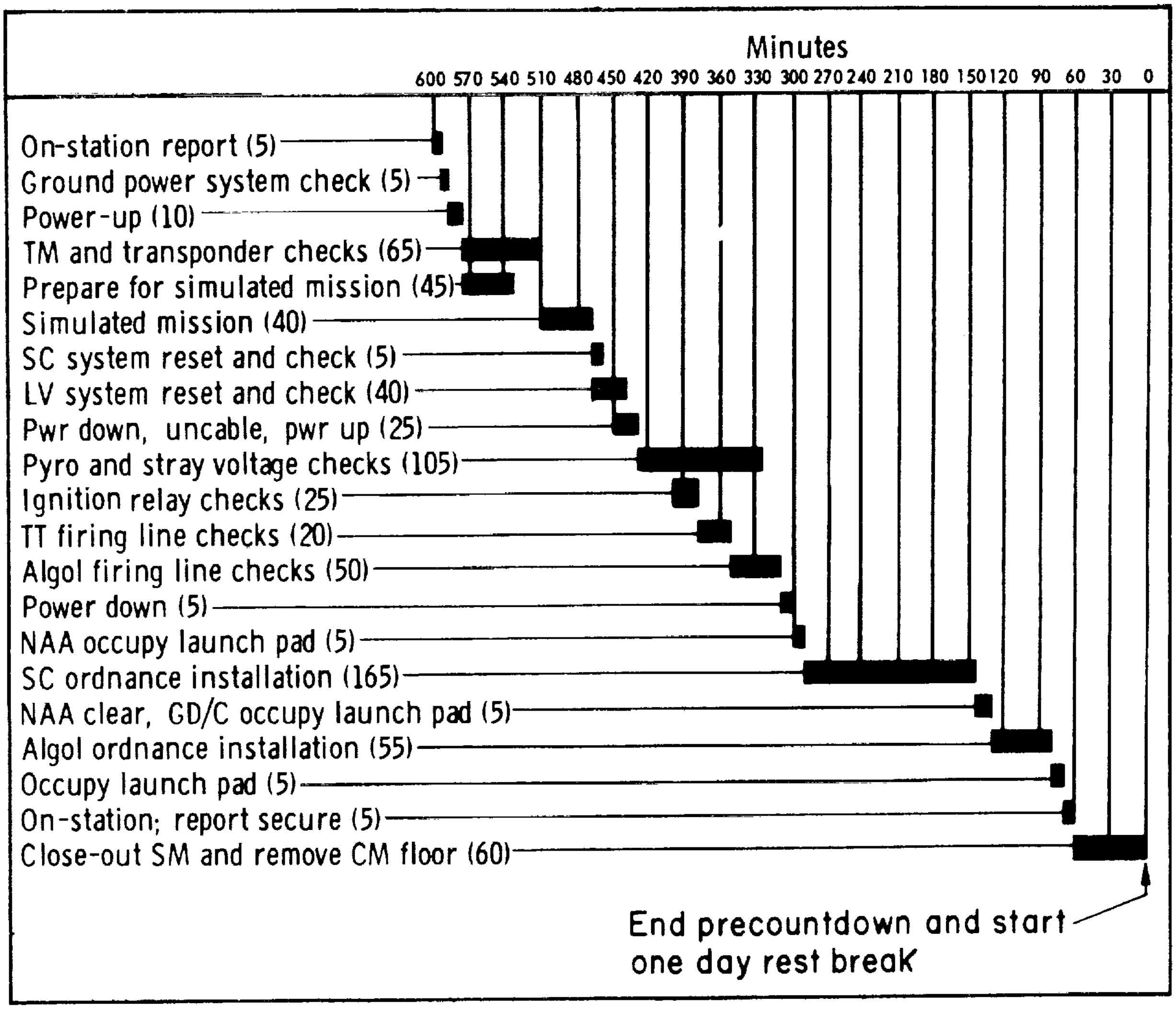


Figure 8-5. - Typical prelaunch test schedules
(a) BP-22 spacecraft precountdown chart

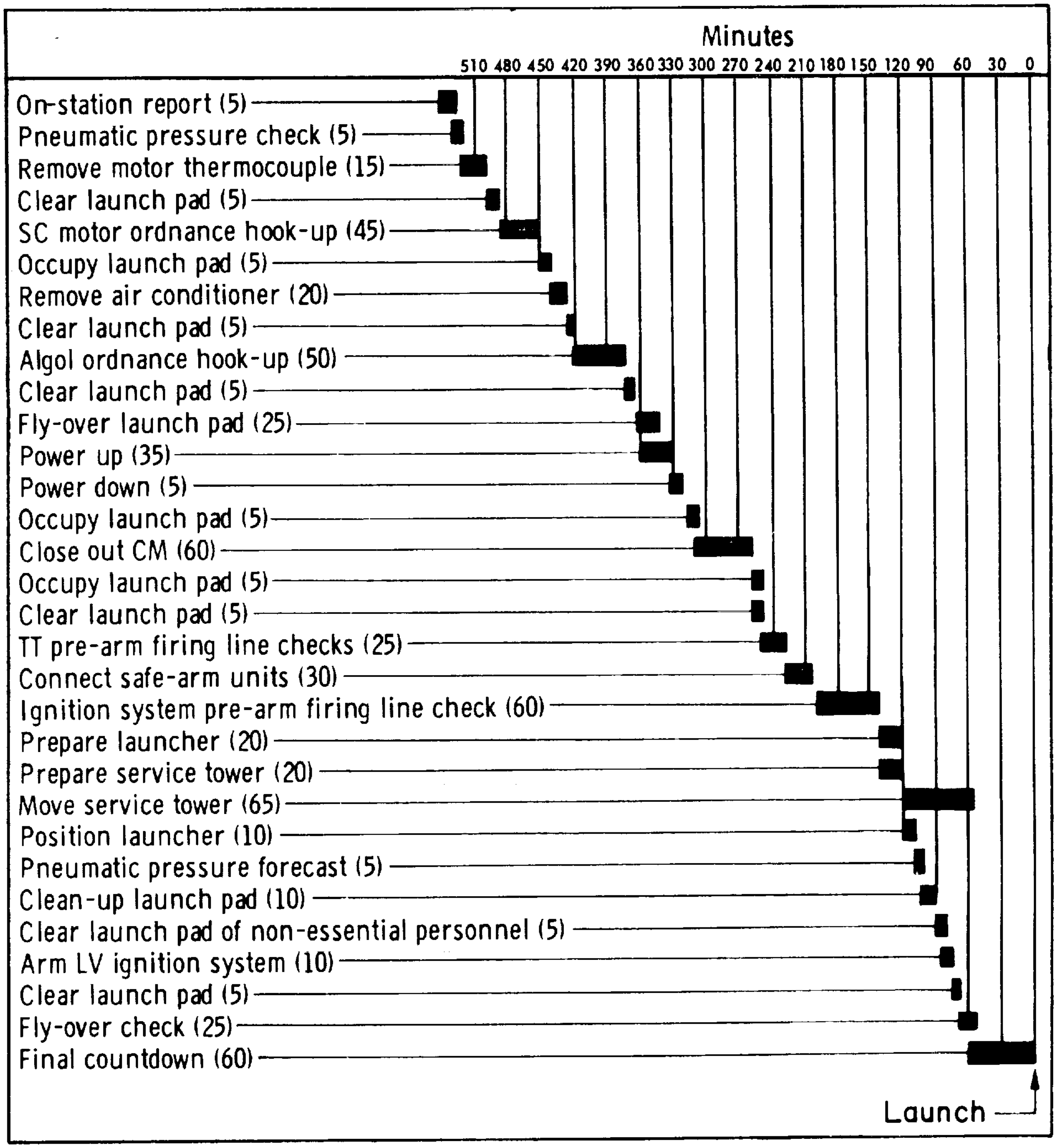


Figure 8-5. - Typical prelaunch test schedules
(b) BP-22 spacecraft countdown chart

9.0 RECOVERY REQUIREMENTS

9.1 Requirements

The primary requirement is the recovery of the command module, tape recorded data,and cameras. Recovery activities will be initiated immediately after impact. Detailed recovery requirements are specified in the Operations Requirements Document 90202, reference 3.

9.2 Recovery Concept

Immediately after impact, the recovery team equipped with the necessary recovery vehicles and specialized personnel will proceed to the impact points and deactivate pyrotechnics. The data tape and cameras will be removed and delivered to NASA; recovered hardware will be returned to the launch site for postflight analysis as required, after which disposition will be determined by the Apollo Spacecraft Program Office.

9.3 Recovery Forces

NASA personnel are responsible for directing White Sands Missile Range (WSMR) ordnance personnel in disarming pyrotechnics not expended during flight and in removing the onboard tape recorder, tape, and cameras. WSMR personnel are responsible for loading and transporting the recovered hardware back to the work area.

9.4 Location Aids

C-band radar beacon, phototheodolite, observation aircraft, and onsite recovery teams will be used in locating the spacecraft.

10.0 WEATHER, PAD SAFETY, AND RANGE SAFETY REQUIREMENTS

10.1 Weather Restraints

Flight restraints due to visibility and/or other weather conditions are specified in the Mission Rules, reference 12.

10.2 Limited Access Areas

The launch pad, blockhouse, and rocket motor assembly areas will be designated limited access areas. Only those personnel required to perform assigned tasks will be permitted access.

Personnel admitted to the limited access area will be required to wear safety helmets, safety shoes, and safety glasses as necessary. An ambulance with crew and a fire truck with crew will be required to stand by during all operations with pyrotechnics, and hospital and medical care will be available if necessary.

10.3 Explosive Control

The rocket motors and explosive devices will be stored by WSMR, until needed, in the hazardous assembly area or at the launch site. Pyrotechnic simulators will be utilized in all checkout procedures whenever possible.

A separate area will be designated for rocket motor buildup. The rocket motors will be transported by WSMR to the launch site as required for vehicle buildup. At all times, except for the checkout procedure, the igniter squibs will have shorting plugs installed and rocket motor case grounded. During vehicle buildup, the service tower will provide lightning protection and the vehicle will be connected directly to an earth ground to prevent static electricity buildup.

10.4 Standard Operating Procedures

Standard Operating Procedures (SOP's) will be prepared as directed in reference 13, and will conform to the requirements of reference 14. These SOP's define safety measures taken during the handling of the rocket motors, installation of igniters, testing of ignition circuits, installation of destruct charges and associated circuitry, utilization of shorting plugs, and final arming of the rocket motors and other pyrotechnic devices.

10.5 Range Safety Support Data

Detailed pyrotechnic, vehicle circuitry, and ordnance handling data will be contained in reference 15.

11.0 DATA HANDLING, ANALYSIS AND REPORTING

11.1 Data Handling and Processing

A data acquisition plan, reference 11, will be issued and will detail plans for the acquisition and distribution of data.

11.1.1 Prelaunch and checkout data.- Test vehicle prelaunch and checkout data will be recorded by the checkout trailer at the site. These data will be displayed on oscillograph recorders, brush-type recorders, and on tabular printouts as specified by the detailed checkout procedures, and used at the test site for analysis. Copies of the magnetic tapes will be made and stored at the base facility to provide additional playback capability at a future date as required. The original magnetic tapes will be transmitted to the Central Data file (ED 14), Houston, after the preliminary analysis has been completed.

11.1.2 Launch telemetry data.- The launch telemetry data will be recorded by at least three WSMR telemetry recording sites. In addition, the NASA telemetry trailer will also record telemetry information from the test vehicle. All stations will record telemetry on magnetic tapes. Copies will be immediately made and forwarded to MSC-Houston and NAA-Downey.

Certain real-time telemetry parameters will be displayed in real-time on oscillographs and pen recorders by participating telemetry stations. The WSMR range telemetry station displays will be utilized for "quick-look" data and analysis at the site by postlaunch analysis team personnel. The NASA telemetry trailer real-time recorder displays will be delivered to the blockhouse for quick-look evaluation by the Test Director and subsequent use of the analysis team. Parameters not displayed in real-time will be recorded on magnetic tape and will be available for playback shortly after conclusion of the flight.

11.1.3 Onboard magnetic tape data.-

11.1.3.1 Prelaunch and checkout: During prelaunch and checkout, data gathered by the onboard tape recorder will be processed and utilized at the test site to support the launch preparation and vehicle checkout phase. These tapes will be forwarded to Central Data file (ED) 14), Houston.

11.1.3.2 Postlaunch: Upon retrieval, the onboard tapes will be forwarded to the NASA TM Van where a verified master and two copies from each master will be made. The verified masters will be forwarded to Houston for data reduction. One tape copy will be forwarded to NAA-Downey and one tape copy will be retained at WSMR.

11.1.4 Range instrumentation.- All range instrument data shall be processed into the required form at WSMR. Copies of these data will be made available to MSC, NAA, and GD/C personnel at the site.

11.1.4.1 Weather data: Weather data, taken 1 hour prior to launch and during the flight, will be supplied by the range in a tabulated form as per section 3 of reference 3. Copies of these data will be made available to MSC, NAA, and GD/C personnel at the site for transmittal to Houston, Downey, and San Diego, respectively.

11.1.4.2 Radar data: Radar plots from each station will include downrange distance versus time and downrange distance versus altitude. Four plot boards will be plotted simultaneously. One plot will be retained at the site to satisfy quick-look data and analysis requirements, the remainder will be distributed to NAA-Downey, GD/C-San Diego, and MSC-Houston. If optical tracking is successful, final radar data will not be required. Otherwise, final radar data will be produced by the range in the coordinate system specified for optical data.

11.1.4.3 Optical data: Optical data will be reduced at WSMR as specified in reference 3. Position and velocity data will be available within 48 hours, attitude data will be available within 15 days. Copies of these data will be made available to MSC, NAA, and GD/C personnel at the site.

11.1.4.4 Sequential film data: Motion picture and still photography sequential film data will be as defined in section 9 of reference 3. The original and four copies will be made available at WSMR within 48 hours of the test. The original and one copy will be transmitted to MSC-Houston, one copy will remain at the site, one copy will be transmitted to NAA-Downey, and one copy shall be transmitted to GD/C-San Diego.

11.2 Calibration Data

The procedure and requirements for handling calibration data will be published at a later date.

11.3 Data Analysis and Test Evaluation

All data, telemetry, onboard tape recorder, radar, theodolite, film, etc., shall be referenced to the "4 inch" vehicle-rise event as zero time. This time, in terms of range time, shall be agreed upon by the range, all contractors, and NASA-MSC personnel within 5 hours of the test and shall be used for all further processing of data.

11.4 Mission Analysis and Reporting

The ASPO Test Division will have overall responsibility for the management of mission analysis and reporting. Contributions to this effort will be required from other organizational elements of MSC located at the launch site and in Houston, Texas. Contributions will also be required from NAA and GD/C. A reporting plan (reference 16), which outlines the analysis procedures and report format and designates the analysis and reporting team charged with the responsibility for implementing the plan, will be prepared by the ASPO Test Division prior to the mission.

The mission results will be reported as described in the following paragraphs:

11.4.1 One-hour report.- The T+1 hour report will be in the form of a telegram prepared at WSMR. This telegram will simply be a notification that the test was completed, approximate test duration, and pertinent test conductor's comments.

11.4.2 Flight status report.- This is a preliminary analysis report published in the form of a telegram approximately 48 hours after flight. This report will be based on quick-look data. It will be written formally by MSC, ASPO, NAA, and GD/C personnel at WSMR under the direction of the ASPO Test Division, and will be forwarded to MSC and NAA management. Additional flight status reports will be written at 24-hour intervals if required.

11.4.3 Postlaunch report.- The postlaunch report will be prepared by NASA and contractor personnel, and will be issued by NASA within 28 calendar days after the launch. The major portion of the analysis and reporting activity will be completed at MSC-Houston.

11.4.4 Contractor spacecraft analysis report.- This report will contain the full spacecraft analysis and will be issued within 30 calendar days after launch by NAA-Downey to ASPO for further disposition.

12.0 GROUND SUPPORT EQUIPMENT

The ground support equipment requirements for the spacecraft are shown in reference 17.

APPENDIX A

ABBREVIATIONS AND SYMBOLS

LIST OF ABBREVIATIONS AND SYMBOLS

The following list includes abbreviations used throughout this document.

ASPO	Apollo Spacecraft Program Office
ATO	Apollo test operations
BP	boilerplate
CM	command module
C/O	checkout
ECS	equipment cooling subsystem
ELS	earth landing subsystem
EPS	electrical power subsystem
GD/C	General Dynamics/Convair
GSE	Ground support equipment
LES	launch escape subsystem
LET	launch escape tower
LEV	launch escape vehicle
LJ II	Little Joe II
LV	launch vehicle
MSC	Manned Spacecraft Center
NAA	North American Aviation
NASA	National Aeronautics and Space Administration
OR	Operations Requirements
PAM	pulse amplitude modulated
PDM	pulse duration modulated
RASPO	Resident-Apollo Spacecraft Program Office

LIST OF ABBREVIATIONS AND SYMBOLS - Continued

RCS	reaction control subsystem
RFWAR	Requirements for Work and Resources
SC	spacecraft
SM	service module
SOP	standard operating procedure
TM	telemetry
WSMR	White Sands Missile Range

APPENDIX B

TERMINOLOGY DEFINITIONS

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TERMINOLOGY DEFINITIONS

1. First-Order Test Objectives.- First-order test objectives comprise the main purpose for conducting the mission. The mission profile will be tailored to optimize the accomplishment of first-order test objectives.

2. Second-Order Test Objectives.- Second-order test objectives consist of those specific test which constrain (e.g. - must be completed satisfactorily prior to) succeeding flights and of Flight Proof Tests of spacecraft equipment. The minimum spacecraft subsystem and instrumentation configuration required for the mission is established by the First and Second Order Test Objectives.

3. Third-Order Test Objectives.- Third-order test objectives are those which support or enhance succeeding Apollo flights, those which supply supplementary data for overall spacecraft evaluation, and those scientific experiments assigned to the mission.

4. Test.- A specially prepared operation, or series of operations, the intent of which is to determine the resultant characteristics of a component or system when operated under specified, controlled conditions. It is characterized as a unique, developmental procedure; once performed successfully, it usually need not be repeated.

5. Proof.- A test, the performance of which produces evidence that a precise design criteria, specification or requirement is met in a manner such that the design of the tested article(s) is considered to be confirmed.

6. Constraint.- The minimum mandatory testing which must be completed satisfactorily prior to the event being considered. A Flight Constraint not completed will hold the mission under consideration. Constraints are "hard" requirements and will not be arbitrarily removed.

7. Subsystem Priorities.-

Primary: Primary subsystems are those subsystems functionally required for the spacecraft to successfully accomplish 1st and 2nd order test objectives. A positive indication of satisfactory operation of primary subsystems must be available prior to launch.

Secondary: Secondary subsystems are those subsystems not functionally required for the spacecraft to complete its planned mission (1st and 2nd order objectives). Malfunction of secondary subsystems will not require a mandatory countdown hold or scrub.

8. Demonstrate.- Denotes the occurrence of an action or an event during a test. The accomplishment of an objective of this type requires a qualitative answer. The answer will be derived through the relation of this action or event to some other known information or occurrence. This category of objective implies a minimum of airborne instrumentation, and/or that the information to be obtained external to the spacecraft.

TERMINOLOGY DEFINITIONS (Concluded)

9. Determine.- Denotes the measurement of performance of any system or subsystem. This category implies a quantitative investigation of overall operation which includes, generally, instrumentation for measuring basic inputs and outputs of the system or subsystem. The information obtained should indicate to what extent the system is operating as designed. The instrumentation should allow performance deficiencies to be isolated to either the system or to the system inputs.

10. Evaluate.- Denotes the measurement of performance of any system or subsystem as well as the performance and/or interaction of its components that are under investigation. The accomplishment of objectives this type required quantitative data on the performance of both the system or sub-systems and its components. The performance levels will then be analyzed for their contribution toward performance of the system. This category will provide the most detailed information of any of these categories.

11. Obtain Data.- This term denotes the gathering of engineering information which is to be measured to augment the general knowledge required in the development of the overall spacecraft. This category may also be used for supplemental investigation such as environmental studies and ground equipment studies. The degree of instrumentation is not implied by this definition.

12. Verify.- To quantitatively demonstrate the safe functioning, achievement of minimum performance, and operational suitability of equipment. (Demonstrate is defined above as Item 8) "Verify", also, implies a quantitative investigation of overall operation which indicates quantitatively that the system does in fact operate as designed, but does not imply that the instrumentation should necessarily be adequate to allow performance deficiencies to be isolated. "Verify", therefore, and insofar as the instrumentation is concerned, would seem to be somewhat more than "demonstrate" but somewhat less than "determine".

REFERENCES

1. "General Test Plan, Research and Development for Project Apollo Spacecraft", Volumes I & V, SID 62-109-1 and 62-109-5. North American Aviation, Space and Information Systems Division (C)

2. "Apollo Test Requirements", NPC 500-10, NASA (U)

3. "Operations Requirements Document 90202", (being written by FTO-WSMR)

4. NASA Apollo Program Working Paper 1132A "Apollo Measurement Requirements, Boilerplate 22, Apollo Mission A-003". (U)

5. "Project Apollo Flight Mission Directive for Mission A-002", NASA Working Paper No. 1138 (C)

6. "Actual Weight and Balance Report, Boilerplate Stack 22", SID 63-143-14, dated March 16, 1965.

7. "Structural Design Criteria", SID 64-183, North American Aviation, Space and Information Systems Division. (C)

8. "Structural Design and Loads Criteria for Test Launch Vehicle and Launcher-Little Joe Project", GD/C 62-278A, General Dynamics/Convair (U)

9. "Apollo Instrumentation Manual for Apollo Mission A-003", (to be published) (U)

10. "Time Format Standards", IRIG 104-60 (U)

11. "Data Acquisition Plan" (to be published by WSMR)

12. "Mission Rules" (to be published by MSC)

13. Annex B WSMR CIR NR 385-2 (U)

14. "ORD Safety Manual", ORD M7-224 (U)

15. "Integrated Range Safety Support Manual", SID 63-679. North American Aviation, Space and Information Systems Division (U)

16. "Reporting Plan for Apollo Mission A-003 (BP-22)", NASA Working Paper No. 1169. (U)

REFERENCES - Concluded

17. "GSE Requirements and Utilizations for Apollo Development Program", SID 62-564. North American Aviation, Space and Information Systems Division. (U)

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